AD NUMBER:	AD0805683
LIMITATION CHANGES	
TO: Approved for public release; distribution is unlimited.	
FROM: Further dissemination only as directed by Federal Aviation Agency, Washington, DC, 20553, 30 Jun 1966, or higher DoD authority.	
AUTHORITY	
FAA notice dtd 2 Jul 1973	

COMMUNICATIONS/NAVIGATION SUBSYSTEM SPECIFICATION

SST

BOEING MODEL 2707

COMMERCIAL
SUPERSONIC TRANSPORT
PROGRAM

PHASE II-C REPORT

D6A10122-1

THE BOEING COMPANY
SUPERSONIC TRANSPORT DIVISION

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

All distribution of this document is controlled. This document may be further distributed by any holder only with specific prior approval of:

Director of Supersonic Transport Development
Federal Aviation Agency
Washington, D. C. 20553

The distribution of this document is limited because it contains technology identifiable with items excluded from export by the Department of State (U. S. Export Control Act of 1949 as amended).

COMMUNICATIONS/NAVIGATION SUBSYSTEM SPECIFICATION

COMPETITIVE DATA

SST

BOEING MODEL 2707

SUPERSONIC TRANSPORT PROGRAM

●

PHASE II-C REPORT
CONTRACT FA-SS-66-5

JUNE 30, 1966

Prepared For
FEDERAL AVIATION AGENCY
Supersonic Transport Development Program

PREPARED BY [illegible]

APPROVED BY Howard G. Barton

APPROVED BY [illegible]

APPROVED BY Fred A. Maxam

D6A10122-1

COMPETITIVE DATA

THE BOEING COMPANY
SUPERSONIC TRANSPORT DIVISION

ISSUE NO. 9

SUBSYSTEM SPECIFICATION

PERFORMANCE/DESIGN
AND PRODUCT CONFIRMATION REQUIREMENTS

COMMUNICATIONS/NAVIGATION
SUBSYSTEM SPECIFICATION
SUPERSONIC TRANSPORT AIRCRAFT
BOEING MODEL 2707

JUNE 30, 1966

D6A10122-1

REVISION RECORD

Revision Date	Pages Revised	Pages Added	Approval

CONTENTS

PART I

GENERAL

CONTENTS (Cont.)

CONTENTS (Cont.)

PART III

NAVIGATION SYSTEM

ILLUSTRATIONS

ILLUSTRATIONS (Cont.)

TABLES

PART I
GENERAL

1.0 SCOPE

Refer to the specific subsystem section.

2.0 APPLICABLE DOCUMENTS

A Military Specification

B Military Standards

MIL-STD-210A	Climatic Extremes for Military	30 November 58
MIL-STD-810A	Environmental Test Methods for Aerospace and Ground Equipment	

C ARINC Characteristics

a.	ARINC No. 410	Mark 2 Standard Frequency Selector System, including Supplement #1	30 April 65
b.	ARINC No. 547	Airborne VHF Navigation Receiver	1 October 61
c.	ARINC No. 521D	Airborne Distance Measuring Equipment, including Supplement #1	21 January 65
d.	ARINC No. 415	Operational and Technical Guidelines on Failure Warning and Functional Tests	15 February 66
e.	ARINC No. 551	Airborne Glide Slope Receiving System	1 March 62
f.	ARINC No. 552	Airborne Radio Altimeter, including Supplement #1	1 August 65
g.	ARINC No. 550	Airborne ADF Systems Mark 2	1 March 62
h.	ARINC No. 533A	Airborne HF SSB/AM System	11 March 66
i.	ARINC No. 546	Airborne VHF Communication Transceiver System	1 October 61

j.	ARINC No. 531	Airborne Selective Calling Unit	28 June 54
k.	ARINC No. 412	Audio Systems	
l.	ARINC No. 532D	Air Traffic Control Transponder	15 May 66
m.	ARINC No. 560	Airborne Passenger Address Amplifier - Draft 3	2 May 66
n.	ARINC No. 557	Airborne Voice Recorder	10 January 64

D Technical Standard Order

a.	FAA TSO-C36b	ILS Localizer Receiving Equipment	15 June 62
b.	FAA TSO-C40a	VOR Radio Receiving Equipment	8 November 59
c.	FAA TSO-C66a	Airborne Distance Measuring Equipment	15 February '65
d.	FAA TSO-C34b	ILS Glide Slope Receiving Equipment	9 May 62
e.	FAA TSO-C41b	LF ADF Receiver	1 March 63
f.	FAA TSO-C35c	Airborne Radio Marker Receiving Equipment	6 April 62
g.	FAA TSO-C31b	HF Radio Communications Transmitting Equipment	3 August 59
h.	FAA TSO-C32b	HF Radio Communications Receiving Equipment	3 August 59
i.	FAA TSO-37b	VHF Radio Communication Transmitting Equipment	14 August 62
j.	FAA TSO-C38b	VHF Radio Communication Receiving Equipment	14 August 62
k.	FAA TSO-C84	Minimum Performance Standards for Cockpit Voice Recorders	14 August 62
l.	FAA TSO-C51a	Aircraft Flight Recorder	29 December 65

E RTCA Papers

a.	RTCA Paper DO-132	Minimum Performance Standards-ILS Glide Slope Receiving Equipment	15 March 66
b.	RTCA Paper 158-61/DO-111	Minimum Performance Standards, Airborne Radio Receiving and Direction Finding Equipment Operating within the Frequency Range of 200 to 415 kH_Z	10 August 61
c.	RTCA Paper 87-54/DO-57A	Minimum Performance Standards, Airborne Marker Receiver Equipment	8 March 62

F General

a.	NASA, Technical Note D-822	Tables of Airspeed, Altitude and Mach Number Based on Latest International Values, for Atmospheric Properties and Physical Constants	August 61
b.	FAR Part 25	Airworthiness Standards, Transport Category Airplanes	14 November 65
c.	FAA A. C. 25.1457-1	Cockpit/Voice Recorder Installation	4 July 65

G Boeing Publications

a.	D6A10064-1	Reliability Analysis Document SST System and Aircraft	5 November 66
b.	D6A10073-1		
c.	D6A17850	Model Specification, Section 1, Aircraft Characteristics and System Requirements	15 November 65
d.	D6A10107-1	Airframe Subsystem Specification	30 June 66

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. Refer to specific subsystem section.

3.1.1.1 Performance Characteristics. Refer to specific subsystem section.

3.1.2 Operability. The allocation of the reliability and maintainability defined herein has been accomplished by analysis and experience and may be revised as long as the overall aircraft requirements as specified in the model specification (D6A17850) are satisfied. For definition of terms used refer to Par. 3.1.2 of D6A10107-1.

3.1.2.1 Reliability. Refer to specific subsystem section.

3.1.2.2 Maintainability. Refer to specific subsystem section.

3.1.2.2.1 Maintenance and Repair Cycle. The scheduled check intervals and the times established for these checks are shown here. All servicing and scheduled inspections shall be fitted within one of these cycles.

Scheduled Check	Time Interval	Elapsed Time
Transit Service	N/A	30 minutes
Turnaround Service	N/A	90 minutes
Daily Check	50 flight hours	1 hour
Intermediate Check	300 flight hours	4 hours
Periodic Check	1,200 flight hours	16 hours
Basis Check	8,400 flight hours	5 days

3.1.2.2.2 Service and Access. Refer to specific subsystem section.

3.1.2.3 Useful Life. The subsystems defined herein shall have a useful life commensurate with that of the aircraft (50,000 hours), assuming normal maintenance and operation of equipment.

3.1.2.4 Environmental. The subsystems defined herein shall perform their intended function under the normal operating conditions defined in Par. 3.1.2.4 of D6A10107-1.

3.1.2.4.1 Pressure Altitude.

a. Pressurized Areas. Minus 1,000 feet to plus 30,000 feet. The equipment shall operate following a sudden decompression from 6,000 feet to 30,000 feet.

b. Unpressurized Areas. Minus 1,000 feet to plus 80,000 feet.

3.1.2.4.2 Operating Temperature.

a. Pressurized Areas. Minus 50°F to plus 120°F continuously with temperatures to 160°F for periods not exceeding 30 minutes.

b. Unpressurized Areas. As specified in subsystem section.

3.1.2.4.3 Storage Temperature. Minus 50°F to plus 160°F.

3.1.2.4.4 Temperature Shock.

a. Pressurized Areas. Minus 50°F to plus 160°F.

b. Unpressurized Areas. As specified in subsystem section.

3.1.2.4.5 Humidity. From 0 to 100 percent relative humidity including conditions of condensation on the equipment in the form of water or frost.

3.1.2.4.6 Vibration. Vibration levels in accordance with subsystem specification D6A10107-1.

3.1.2.4.7 Shock. 18 impact shocks of 15g's, consisting of 3 shocks in opposite directions along each of 3 mutually perpendicular axes, each shock impulse having a time duration of 11 ±1 milliseconds.

3.1.2.4.8 Sand and Dust. As defined in MIL-STD-810A Method 510-1.

3.1.2.4.9 Salt Atmosphere. Salt spray containing 0.003 to 1.5 ppm by weight of sea salt from sea level to 5,000 feet.

3.1.2.4.10 Icing. As specified in subsystem section.

3.1.2.4.11 Acoustics. Noise levels up to 135db.

3.1.2.5. Human Performance. All components which form a part of a functional man-machine interface shall be designed in accordance with human engineering principles outlined in D6A10109-1, Flight Deck Subsystem Specification.

3.1.2.6 Safety.

3.1.2.6.1 Flight Safety. The design shall be such that no combination of failure identified in the "failure modes and effects analyses" (see D6A10064-1 for analyses) shall result in the existence of a condition of catastrophic hazard.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements. Refer to specific subsystem section.

3.2.1.1 Schematic Arrangement. Refer to specific subsystem section.

3.2.1.2 Detailed Interface Definition. Refer to specific subsystem section.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. Refer to specific subsystem section.

3.3.2 Selection of Specification and Standards. Reference Airframe Subsystem Specification D6A10107-1.

3.3.3 Materials, Parts, and Processes. Reference Airframe Subsystem Specification D6A10107-1.

3.3.4 Standard and Commercial Parts. Reference Airframe Subsystem Specification D6A10107-1.

3.3.5 Moisture and Fungus Resistance. Reference Airframe Subsystem Specification D6A10107-1.

3.3.6 Corrosion of Metal Parts. Reference Airframe Subsystem Specification D6A10107-1.

3.3.7 Interchangeability and Replacement. Reference Airframe Subsystem Specification D6A10107-1.

3.3.8 Workmanship. Reference Airframe Subsystem Specification D6A10107-1.

3.3.9 Electromagnetic Interference. Reference Airframe Subsystem Specification D6A10107-1.

3.3.10 Identification and Marking. Reference Airframe Subsystem Specification D6A10107-1.

3.3.11 Storage. Reference Airframe Subsystem Specification.

4.0 QUALITY ASSURANCE PROVISIONS

This paragraph specifies the requirements and methods of verification for each performance requirement specified in Parts II and III of this specification.

4.1 ENGINEERING TEST AND EVALUATION. Engineering Test and Evaluation (ET and E) tests are performed as an integral part of the development process to acquire data to support the design process and to verify subsystem performance with respect to flight safety, normal operation, and abnormal operation.

4.2 PRELIMINARY QUALIFICATION TESTS. Preliminary qualification tests are not applicable to the Communication/Navigation system.

4.3 FORMAL QUALIFICATION TESTS. These tests are conducted to verify that all requirements of Parts II and III are satisfied. Accomplishment of verification constitutes acceptance of design and development.

4.3.1 Inspections.

a. Review drawings and wiring diagrams to establish compliance with design requirements.

b. The design features shall be verified by development data and airplane inspections.

c. System displays equipment malfunctions, and monitoring provisions shall be verified by inspection and operational demonstrations.

4.3.2 Analysis. An analysis to verify system compliance with design requirements shall be conducted. The analysis shall include a review of performance and functional characteristics under specified environmental conditions and all aircraft flight modes. A failure mode and effects analysis shall be performed to verify satisfactory operational characteristics during normal and abnormal system conditions.

4.3.2.1 Maintainability. The maintainability requirements under Par. 3.1.2.2 represent the mature system operated in representative scheduled airline revenue service. Projections of the requirements will be verified by analysis of data acquired as a result of, and in conjunction with, mockup evaluations, qualification tests, developmental tests, engineering airplane ground tests, and flight tests. All activities involving scheduled checks, repairs and servicing of LRU will be observed and data recorded. The suitability of service and access provisions will be determined by observation of technicians performing maintenance and servicing tasks on the subsystem.

4.3.2.2 Useful Life. Analytical review of applicable design, tests, and service data shall be provided to justify useful life requirements of Par. 3.1.2.3.

4.3.2.3 Environmental. Verification shall be based on analytical review of development tests and component qualification tests to substantiate the requirements of Paragraphs 3.1.2.4 through 3.1.2.4.11.

4.3.2.4 Human Performance. Refer to Flight Deck Subsystem Specification D6A10109-1.

4.3.2.5 Safety. The safety requirements identified in Par. 3.1.2.6 shall be verified analytically by the identification of compensating provisions for each failure mode defined in the failure mode effect and criticality analysis.

4.3.3 Demonstrations. There are no scheduled demonstration tests. The characteristics and adequacy of the subsystems shall be demonstrated during ground tests.

4.3.4 Tests. Functional tests shall be conducted on components and equipment as necessary to verify operational and nonoperational compatibility with temperature, pressure, and vibration. EMI tests will be conducted at the component level on the prototype airplane to verify the requirements specified in D610107-1 (Airframe Subsystem Specification).

a. Subsystem Integration Tests. Integrated system operation and interface provisions of the subsystems shall be developed and verified in the Comm/Nav test rig.

b. Airplane Tests. Capability for Comm/Nav system operation shall be demonstrated by ground and flight tests. Refer to specific subsystem sections for detailed testing.

4.4 RELIABILITY TEST AND ANALYSIS. The reliability requirements of Par. 3.1.2.1 represent the mature system operated in representative scheduled airline revenue service. Inasmuch as the tests and data specified in Par. 4.3.4 are limited and the hardware may be of a prototype nature, compliance with the requirements of Par. 3.1.2.1 will be accomplished as follows:

4.4.1 Reliability Tests. Tests specifically designed to verify the reliability of the subsystem shall not be conducted. Data obtained from tests conducted under Par. 4.3.4 shall be applied to the reliability analysis specified in Par. 4.4.2 extrapolated to anticipated airlines operational conditions.

4.4.2 Reliability Analysis. A reliability analysis shall be performed to demonstrate that the requirements of Par. 3.1.2.1 can be achieved. This shall be accomplished as follows:

a. A growth curve shall be established to base the target reliability levels projected by the end of Phase III.

b. Design data and test results will be applied to a reliability analysis model incorporating:

   (1) Block diagrams summarizing the logical relationships between component success/failure and system success/failure.

   (2) A mathematical reliability model derived from (1) and incorporating minimum equipment requirements for continued flight.

   (3) A mathematical reliability model simulating typical airline operations and routes.

c. Comparison shall be provided with the Phase III targets and the results extrapolated to determine expectation of achieving the requirements of Par. 3.1.2.1 in airlines operation.

QUALITY ASSURANCE PROVISIONS
CROSS REFERENCE INDEX

4.0 QUALITY ASSURANCE PROVISIONS.

4.1 ENGINEERING TEST AND EVALUATION.

4.2 PRELIMINARY QUALIFICATION TESTS.

4.3 FORMAL QUALIFICSTION TESTS.

- 4.3.1 Inspections.
- 4.3.2 Analysis.
- 4.3.3 Demonstrations.
- 4.3.4 Tests.

4.4 RELIABILITY TEST AND ANALYSIS.

SECTION 3.0 PARAGRAPH	SECTION 4.0 PARAGRAPH
3.1.1	4.3
3.1.1.1	4.3
3.1.2	NA
3.1.2.1	4.4
3.1.2.2	4.3
3.1.2.2.1	4.1
3.1.2.2.2	4.3
3.1.2.3	NA
3.1.2.4	4.3
3.1.2.5	NA
3.1.2.6	NA
3.2	NA
3.2.1	NA
3.2.1.1	NA
3.2.1.2	4.3
3.3	NA
3.3.1	4.1
3.3.2	4.3
3.3.3	4.3
3.3.4	4.3
3.3.5	4.3
3.3.6	4.3
3.3.7	4.3
3.3.8	4.3
3.3.9	4.3
3.3.10	4.3
3.3.11	4.3

5.0 ADDITIONAL DATA

PART II COMMUNICATION SYSTEM
SECTION A HF COMMUNICATION

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test, and qualification of equipment identified as the High Frequency (HF) Communication Subsystem as applied to the prototype airplane. This subsystem shall provide two-way long-range communication coverage with airplanes and ground stations.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. HF Communications Systems shall provide two-way communications coverage beyond line of sight in the 2- to 30-mHz frequency band while utilizing SSB and compatible AM type signals as defined by ARINC Characteristic 533A. A visual and/or aural warning at selected ground initiated contact shall be provided to the flight crew.

3.1.1.1 Performance Characteristics. The system shall provide omnidirectional communication and perform in accordance with FAA TSO-C31b, FAA TSO-C32b, and ARINC Characteristic 533A.

With favorable propagation conditions, communications shall be possible throughout the HF spectrum when utilizing ARINC or other FAA approved ground stations.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the High Frequency (HF) Communication Subsystem shall not exceed 1.68 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the HF Communication Subsystem shall not exceed a mean expenditure of 67.8 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. Servicing shall be limited to component replacement.

Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The HF antenna structure shall be designed to withstand without damage or impairment of performance the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The HF communication equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITIONS.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 1.

3.2.1.2 Detailed Interface Definition. The HF Communications Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The system shall interface with the airplane interphone subsystem via microphone audio and supply circuits, microphone control circuits, and receive

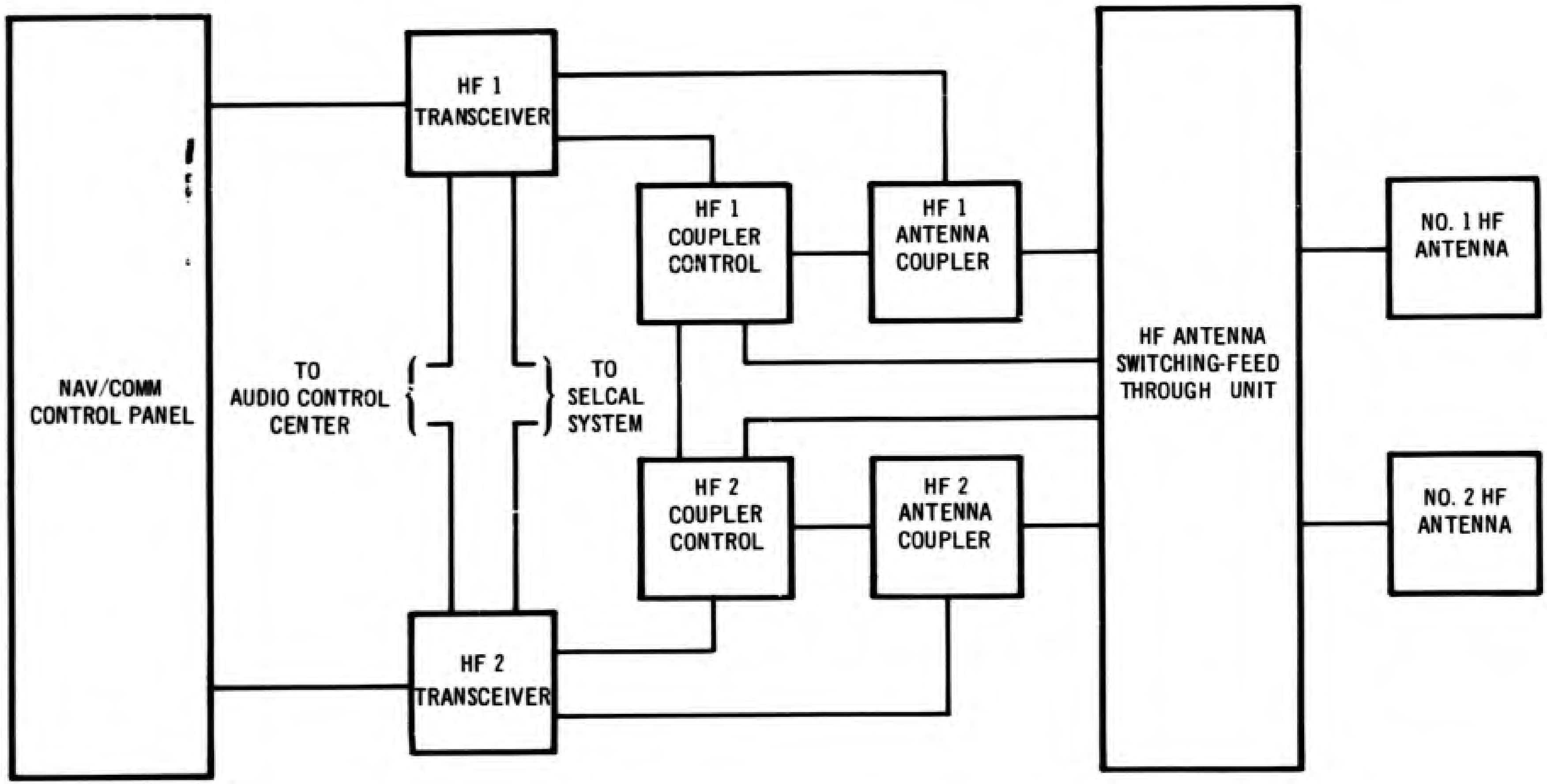


Figure 1. Communications System

audio circuits. The system shall interface with the airplane SELCAL System via an isolated receive audio output from the HF system.

The power requirements shall not be greater than that specified in the following:

R/T unit, receiver	700 watts
Transmit SSB	1,700 watts
Transmit AM	2,000 watts

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The HF Communications set shall consist of units per ARINC 533A.

The weight of the system shall not exceed 384.4 pounds.

The equipment shall consist of the following units: two receiver-transmitters (R/T) located in the forward equipment racks, two control panels located in the flight deck area and accessible to both pilots, two antenna coupler controls located in an associated switching unit, and two antennas.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

a. Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a free-space antenna range.

The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer t Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify that the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the flight test demonstrations which verify the design and performance requirements as specified in Par. 3.0 of this specification.

4.3.4.3.1 An ARINC HF ground station or other FAA approved HF ground installation shall be used as the HF ground terminal for the flight test.

4.3.4.3.2 One HF radio contact (two-way voice communication) shall be established prior to takeoff. Satisfactory reception will be determined by the operating personnel at both ends of the radio link. Record data (frequency, station called, time of day, quality of reception, etc.).

4.3.4.3.3 When at cruising altitude, establish communication with a line-of-sight ground station. When communication is established, fly a complete circle while maintaining communications to determine if any holes exist in the antenna pattern. Model, antenna range, patterns will be compared to any holes that are found to exist.

4.3.4.3.4 At least three HF frequencies, spread throughout the band of frequencies in which the equipment is operable, shall be utilized to establish beyond line-of-sight communications with authorized ground stations. All three contacts shall be readable.

4.3.4.3.5 All flight systems shall be monitored to determine if the HF system is causing any interference. No degradation of any flight system is allowed.

4.3.4.3.6 The HF system shall be monitored to determine if any flight system is causing interference in the HF system. No degradation is allowed.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART II COMMUNICATION SYSTEM
SECTION B VHF COMMUNICATION

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test, and qualification of equipment identified as the VHF Communications Subsystem as applied to the prototype airplane. This subsystem shall provide two-way voice communication in the 118- to 136-mHz frequency band.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. A dual VHF communications subsystem shall provide line-of-sight two-way communications coverage in the 118- to 136-mHz frequency band. Provisions shall be included for a third system. A visual and/or aural warning of selected ground initiated contact shall be provided to the flight crew. The transceivers shall be capable of angle modulation for future implementation of the proposed Aeronautical Mobile Satellite Communications Service.

3.1.1.1 Performance Characteristics. The VHF Communications Subsystem shall perform in accordance with ARINC Characteristics 546, FAA TSO-C37b, and FAA TSO-C38b.

The VHF Communications Subsystem shall conform to requirements of FAR Part 25.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the VHF Communications Subsystem shall not exceed 0.36 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the subsystem is assumed. Definition of flight turnbacks or deviations is in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours shall not exceed a mean expenditure of 50.9 direct maintenance manhours per 1,000 flight hours on and off the airplane, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environmental.

3.1.2.4.1 The VHF Communications antenna structure shall be designed to withstand, without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperatures. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fuel Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The VHF Communications equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 2.

3.2.1.2 Detailed Interface Definition. The VHF Communications Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The system shall interface with the airplane interphone subsystem via microphone audio and supply circuits, microphone control circuits and receive audio circuits. The system shall interface with the airplane SELCAL System via an isolated receive audio output from the VHF Subsystem.

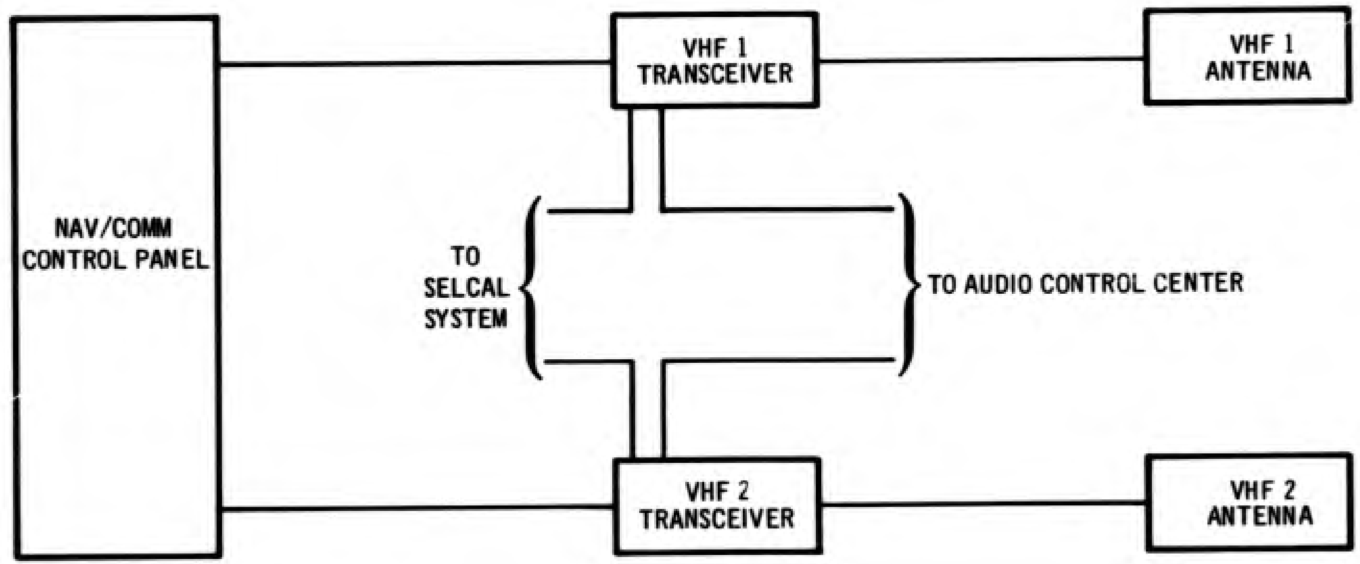


Figure 2. Communication System

The power requirements of the VHF transceiver shall not be greater than that specified below:

Receive	4 amperes at 27.5 volts direct current
Transmit	13 amperes at 27.5 volts direct current

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The VHF Communications set shall consist of units per ARINC 546. The weight of the transceiver shall not exceed 74.8 pounds.

The equipment shall consist of the following units: two receiver-transmitters (R/T) located in the forward equipment racks, two control panels, located in the flight deck area (accessible to both pilots), and two antennas.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TEST AND EVALUATION. Pattern and radiation tests shall be performed on scale models of the airplane to establish the optimum antenna configuration and locations. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

4.1.1.1 Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

4.1.1.2 The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4, Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify subsystem installation performance in the airplane environment as specified in Par. 4.3.4 Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements as specified in Par. 3.0 of this section.

4.3.4.3.1 An FAA-approved VHF ground facility shall be used for the flight test.

4.3.4.3.2 Two-way voice communications shall be established with the VHF ground facility using appropriate frequencies. Loud-and-clear reception, as determined by the operating personnel at both ends of the link shall result from all radio contacts. At least one radio contact shall be established for each of the airplane operational conditions.

4.3.4.3.3 At least three different VHF frequencies shall be used during the flight. The frequencies will be spread throughout the band of frequencies the equipment is capable of to the extent limited only by available ground station frequencies.

4.3.4.3.4 At least one radio contact shall be made near the outer limit of the line-of-sight distance to the VHF radio ground facilities.

4.3.4.3.5 Operational characteristics resulting from antenna patterns shall be observed by placing any antenna pattern null, falling within a cruise-bank angle, in the line-of-sight direction to the ground station while flying at a slant range of between 50 to 150 nmi from the ground station at cruise altitude.

4.3.4.3.6 All flight systems shall be observed for interference caused by the system under test. No degradation of any flight system, because of operation of the system under test, shall exist.

4.3.4.3.7 During the entire flight the system under test shall be observed for interference from any airplane system. No degradation of the system under test, because of operation of any airplane system, shall exist.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART II COMMUNICATION SYSTEM
SECTION C SATELLITE COMMUNICATION

To be added at a later date.

Figure 3 to be added later.

1.0 SCOPE

This section of the specification establishes the requirement for performance, design, test, and qualification of equipment identified as the Selective Calling (SELCAL) Subsystem as applied to the prototype airplane. This subsystem shall provide continuous visual and aural alerting indication when properly triggered.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. A dual-channel SELCAL system shall provide continuous visual and aural alerting indication when triggered by a properly coded audio signal from the VHF communications system or the HF communications system.

3.1.1.1 Performance Characteristics. The alerting devices shall operate continuously until manually reset. The performance of the system shall conform to performance characteristics of the decoder unit as specified in ARINC Characteristic 531.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I

3.1.2.1 Reliability. Flight turnbacks or deviations due to malfunction of the SELCAL subsystem shall not exceed 0.04 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75 hours duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the SELCAL subsystem shall not exceed a mean expenditure of 3.3 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The SELCAL subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment. The SELCAL equipment structure shall be designed to withstand without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 4.

3.2.1.2 Detailed Interface Definitions. The SELCAL Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The power requirements of the decoder unit shall not exceed 0.3 amperes at 27.5-volt direct current for a two-channel unit exclusive of cockpit signal devices.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The SELCAL decoder unit shall consist of units per ARINC 531.

The weight of the SELCAL subsystem shall not exceed 15.1 pounds.

3.3.2 through 3.3.11 — See Part I of this specification

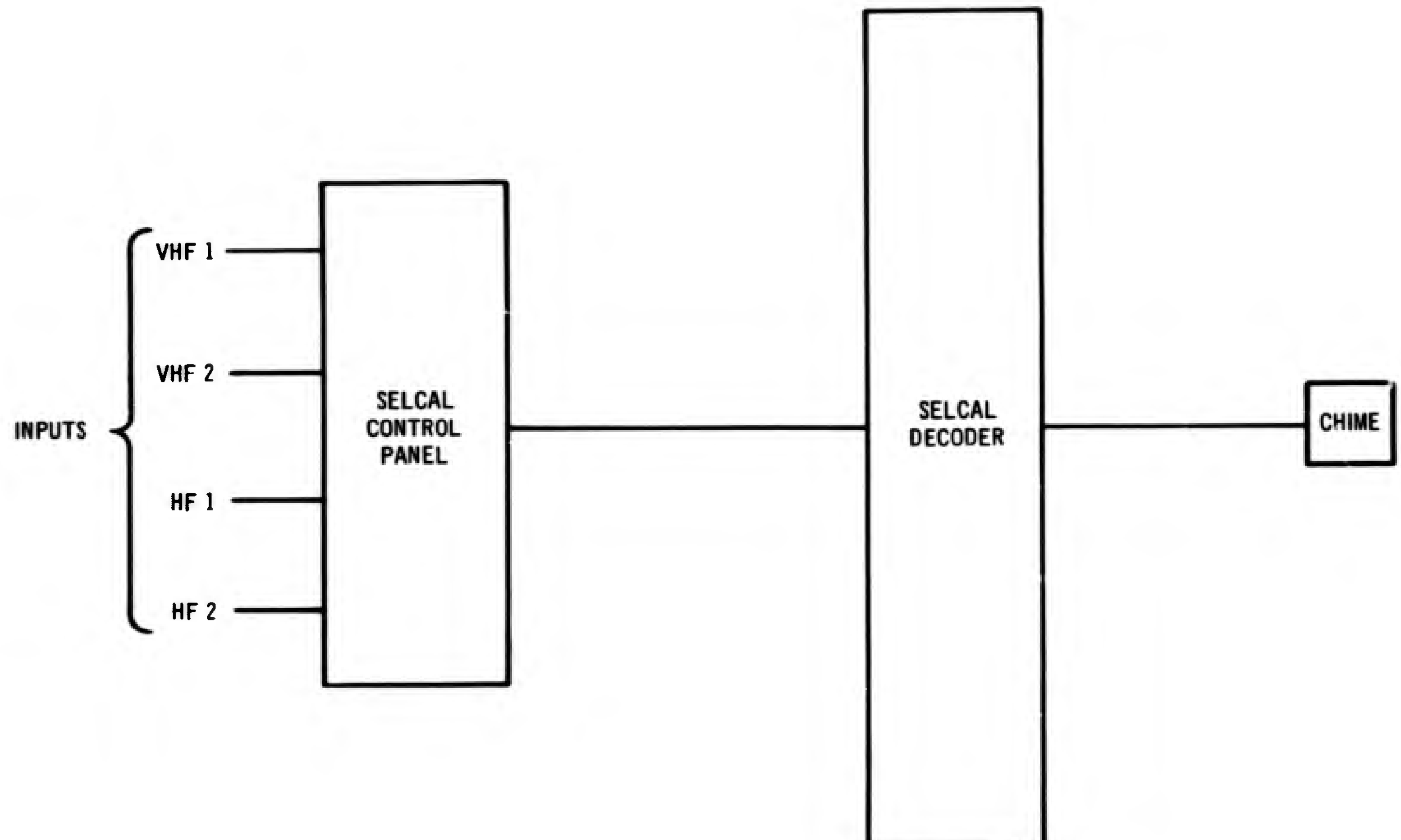


Figure 4. Selective Calling System

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TEST. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA - TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4.

4.3.4.3 Airplane Flight Test. Airplane flight tests shall be conducted to demonstrate operation of the SELCAL System when triggering from the HF communications system and the VHF communications system for all normal airplane operational conditions. Airplane operational conditions shall include:

a. Taxi
b. Climbout
c. Subsonic hold
d. Supersonic cruise
e. Letdown

The visual and aural alerting signalling shall be activated by each triggering.

The SELCAL system and all other systems shall be observed for any indication of interference resulting from the SELCAL system or interference in the SELCAL system due to any other systems.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART II COMMUNICATION SYSTEM
SECTION E INTERPHONE

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test and qualification of equipment identified as the Interphone Subsystem as applied to the prototype airplane. This subsystem shall provide communication among flight crew personnel and between flight crew personnel and ground crew.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The interphone system shall be designed to permit communications between flight crew members and ground crew through a service interphone switch, and shall be a part of the air-to-ground communications system.

3.1.1.1 Performance Characteristics. The system shall perform in accordance with ARINC Characteristic 412.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Interphone Subsystem shall not exceed 2.40 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Interphone Subsystem shall not exceed a mean expenditure of 18.4 direct-maintenance man-hours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The Interphone subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environmental. Refer to Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Figs. 5 and 6.

3.2.1.2 Detailed Interface Definition. The Interphone Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The system shall provide receive audio, transmit audio, and transmit controls circuits, as required, to the following airplane subsystems:

a. VHF Communications Subsystem
b. HF Communications Subsystem
c. Passenger Address Subsystem
d. VOR Navigation Subsystem
e. DME Navigation Subsystem
f. ADF Navigation Subsystem

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The Interphone Subsystem shall consist of units per ARINC 412.

The weight of the system shall not exceed 71.8 pounds.

Audio-selector panels shall be located at the pilots', flight engineer's, and forward observer's stations.

The interphone panels shall be located at the cabin attendants stations, electronics equipment areas, and ground service stations. The interphone amplifier or audio control unit shall be located in the forward electronics equipment area. Handsets shall be installed at the forward and aft cabin attendants stations.

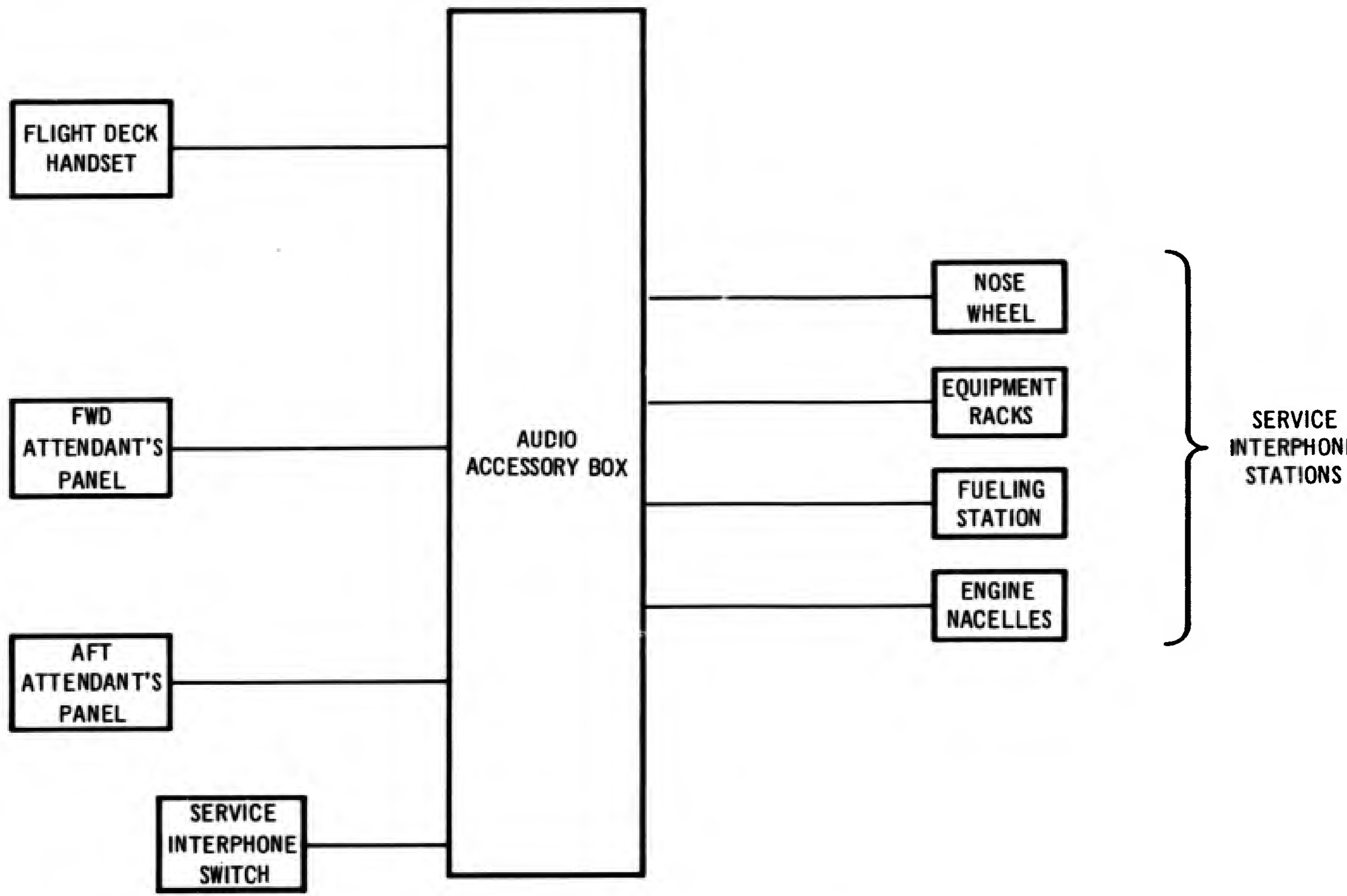


Figure 5. Cabin-Service Interphone System—(Crew-to-Crew)

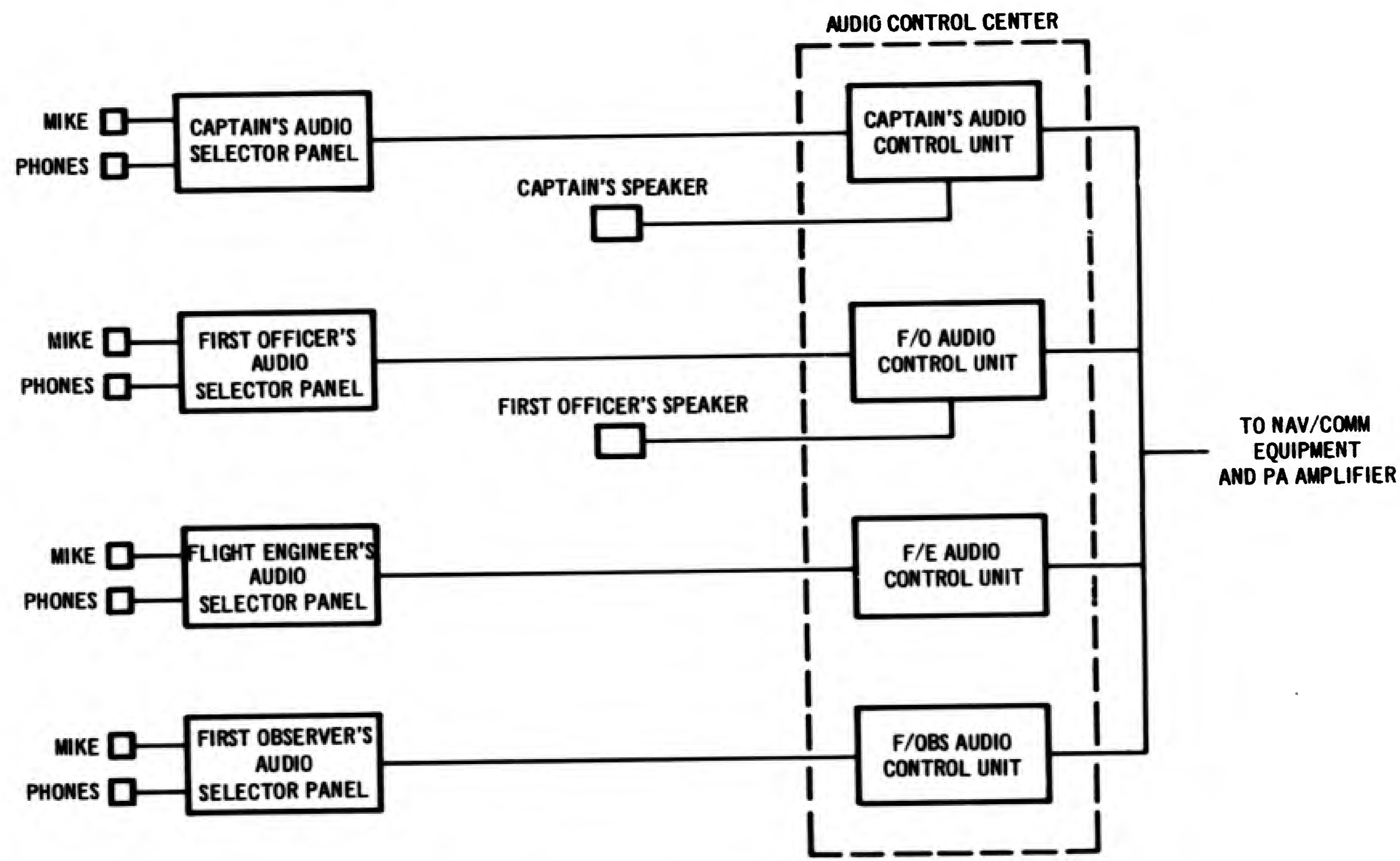


Figure 6. Flight Interphone System – (Crew-to-Ground)

Headphones, boom microphones, oxygen mask microphones, and storage provisions shall be provided at the pilots', flight engineer's, and forward observer's stations.

Loudspeakers, audible at the pilots' seated positions shall be installed in the flight deck.

3.3.2 through 3.3.11	See Part 1 of this specification

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3. of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.2 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Tests. Demonstration of all operational modes of the interphone system for all normal airplane operational modes shall be performed. Airplane operational modes to include: (a) taxi, (b) takeoff, (c) climbout, (d) subsonic hold, (e) supersonic cruise, and (f) letdown.

4.3.4.3.1 Communications shall be intelligible from each interphone station to all other interphone stations.

4.3.4.3.2 The interphone system and all other airplane systems shall be observed for any indication of interference due to the interphone system or interference in the interphone system due to any other system.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.3.4 of Part I.

PART II COMMUNICATION SYSTEM
SECTION F ATC TRANSPONDER

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test and qualification of equipment identified as the Airways Traffic Control (ATC) Subsystem as applied to the prototype airplane. This subsystem shall transmit information pertaining to both the altitude and identity of the airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The dual ATC systems shall provide positive aircraft location within the coverage area, general flight information, altitude information, and discreet airframe identity to the air traffic controller. One system shall operate with the other in standby.

The airborne transponder shall have a digital altitude input and a suppression input/output as specified in ARINC Characteristic 532D.

3.1.1.1 Performance Characteristics. The independent systems shall operate at a received frequency of 1,030 mHz, at a transmitted frequency of 1090 mHz, and to a range of 200 nmi. Each transponder shall be able to reply to an interrogation in any of the four modes (modes A, B, C and D) and will automatically reply to altitude interrogation.

The transponder shall have the capability of employing two-pulse and three-pulse sidelobe suppression.

Each ATC system shall be capable of replying to interrogations at all azimuth angles and up to a 10° bank angle for the full 200-nmi range.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations due to malfunction of the ATC subsystem shall not exceed 0.77 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75 hours duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the ATC subsystem shall not exceed a mean expenditure of 46.4 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The ATC subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The ATC transponder antenna structure shall be designed to withstand without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except for the following modifications:

a. Operating Temperatures. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fuel Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The ATC transponder equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 7.

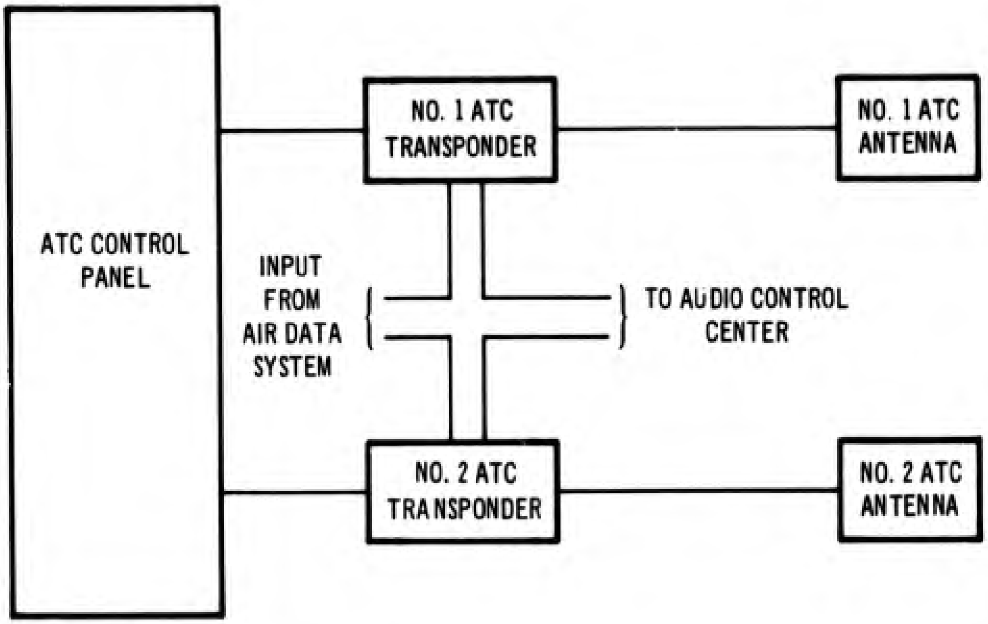


Figure 7. ATC Transponder System

3.2.1.2 Detailed Interface Definition. The ATC system shall provide interface with appropriate ARINC and FAA ground systems.

The power requirements of the transponder shall not exceed 64 VA, 0.84 power factor or equivalent 27.5-volt power.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The Interphone System shall consist of units per ARINC 412.

The weight of the system shall not exceed 71.8 pounds.

Audio-selector panels shall be located at the pilots', flight engineer's, and forward observer's stations.

The interphone panels shall be located at the cabin attendants stations, electronics equipment areas, and ground service stations. The interphone amplifier or audio control unit shall be located in the forward electronics equipment area. Handsets shall be installed at the forward and aft cabin attendants stations.

Headphones, boom microphones, oxygen mask microphones, and storage provisions shall be provided at the pilots', flight engineer's and forward observer's stations.

Loudspeakers, audible at the pilots' seated positions shall be installed in the flight deck.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING AND EVALUATION TESTS. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility, with the following exceptions:

a. Antenna Radiation Pattern

The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

The optimum antenna location shall be substantiated by testing on the same model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.1 of Part I.

4.3.3 Demonstration. Refer to Par 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4, Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA - TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4, Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4, Part I.

4.3.4.3 Airplane Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements specified in Par. 3.0 of this section.

4.3.4.3.1 Beginning at a distance of 5 to 10 nmi from a radar facility and at an altitude of 5,000 feet above the facility, fly the airplane inbound and over the radar facility. At a distance of 5 to 10 nmi beyond the radar facility, fly the airplane at its maximum normal rate of climb to an attitude of 35,000 feet or to the airplanes' maximum certified altitude.

4.3.4.3.2 At an altitude of 35,000 feet, or at the airplanes' maximum certified altitude, fly outbound from the station maintaining a relative heading to the station of 180 ±5° to a distance of at least 160 nmi.

1.0 SCOPE

This section of the specification establishes the requirement for performance, design, test, and qualification of equipment identified as the Passenger Address Subsystem (PA) as applied to the prototype airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The PA system shall provide announcements, emergency pretaped messages and background music in the passenger cabin.

3.1.1.1 Performance Characteristics. The system shall perform in accordance with ARINC Characteristics 412 and 560.

The pilots shall have priority over the cabin attendant use of the PA system.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the PA subsystem shall not exceed 0.07 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the PA subsystem shall not exceed a mean expenditure of 10.7 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The PA subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment. The PA system shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Figure 8.

3.2.1.2 Detailed Interface Definition. The system shall provide interface with appropriate ARINC and FAA ground systems. The power requirements shall not exceed 25 VA at 115-volt, 400 cps.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The PA amplifier shall consist of units per ARINC 412 and 560.

The weight of the amplifier shall not exceed 414.4 pounds. A switch controlling priority between the cabin attendants shall be installed on the forward attendant panel.

Loudspeakers in the passenger cabin shall be located as necessary to provide essentially uniform sound distribution. Supplementary loudspeakers shall be installed in each lavatory. Cabin-attendant initiated PA announcements may be monitored by the flight crew through the flight deck interphone loudspeakers (separate flight deck PA speaker optional).

The PA amplifier and tape reproducer(s) shall be located in the forward electronic equipment racks.

3.3.2 through 3.3.11 } See Part I of this specification.

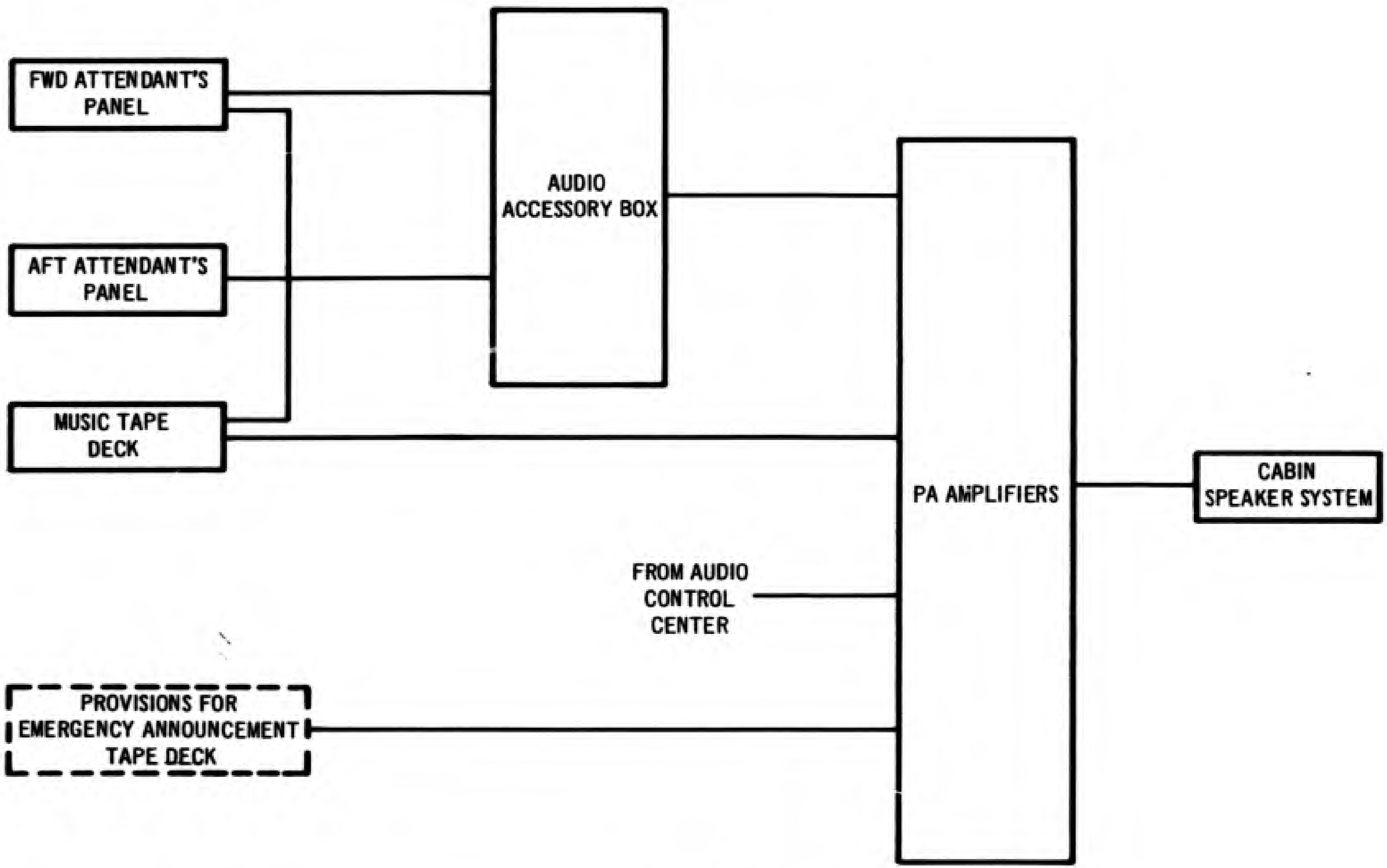


Figure 8. PA System

D6A10122-1

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify that the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.5 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.6 of Part I.

4.3.4.3 Airplane Flight Tests. Demonstration of the PA system for all normal airplane operational configurations shall be performed.

4.3.4.3.1 Airplane operational conditions to include: (a) taxi, (b) takeoff, (c) climbout, (d) subsonic hold, (e) supersonic cruise, and (f) letdown.

4.3.4.3.2 Sound level measurements and listener intelligibility comments shall be recorded from various locations in the passenger cabin for a normal voice input to the system.

4.3.4.3.3 The PA system and all other airplane systems shall be observed for any indication of interference resulting from the PA system or interference in the PA system because of any other system.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART II COMMUNICATION SYSTEM
SECTION H COCKPIT VOICE RECORDER (CVR)

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test and qualification of equipment identified as Cockpit Voice Recorder (CVR) Subsystem as applied to the prototype airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The CVR system shall record voice communications by the flight crew through the interphone system and a cockpit area microphone in a manner that may provide data for accident investigation.

3.1.1.1 Performance Characteristics. The system shall perform in accordance with ARINC Characteristic 557, FAA TSO-C84, FAA A.C. 25.1457-1, and FAR Part 25.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the CVR subsystem shall not exceed 0.01 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the CVR subsystem shall not exceed a mean expenditure of 3.4 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The CVR subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment. The CVR system shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.6 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 9.

3.2.1.2 Detailed Interface Definition. The system shall provide interface with appropriate ARINC and FAA ground systems.

The power requirements shall not exceed 25 VA at 115-volts, 400 cps.

The cockpit voice recorder shall provide for the pickup spoken words of all flight deck personnel via an area microphone located in the flight deck area.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The weight of the voice recorder shall not exceed 55.0 pounds. The recorder unit shall be installed in the aft cargo area, and meets the requirements of TSO-C51a. A microphone and control panel shall be installed to be accessible to both pilots and the flight engineer.

3.3.2 through 3.3.11 } See Part I of this specification.

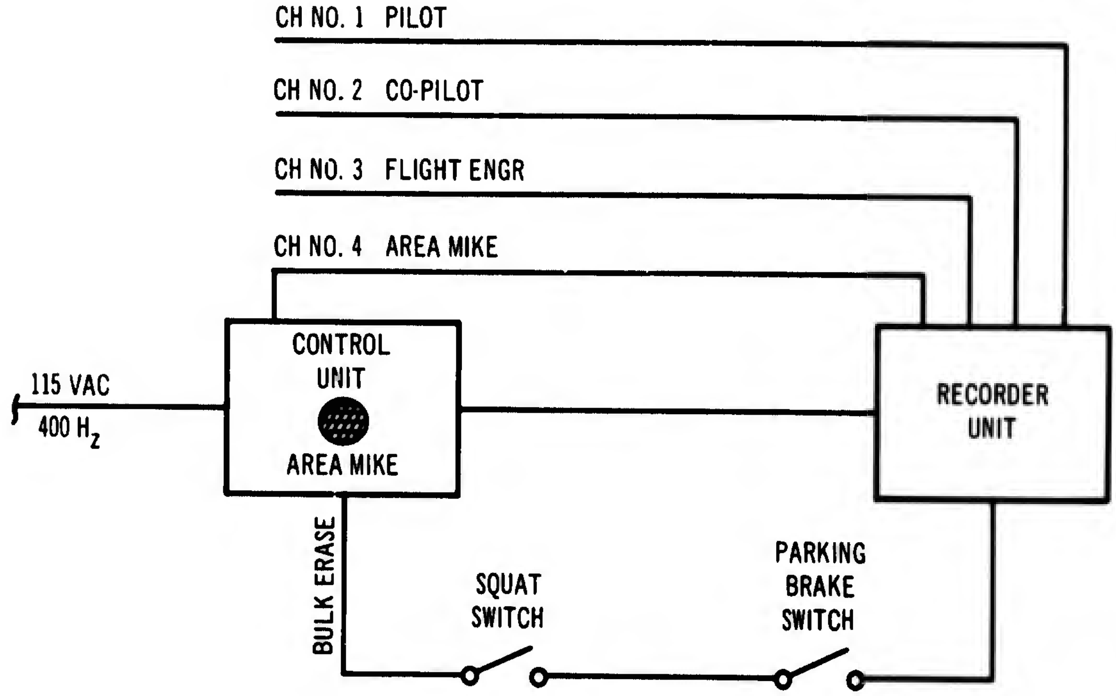


Figure 9. Airborne Voice Recorder

D6A10122-1

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Refer to Par. 4.3.4 of Part I.

4.4 RELIABILITY TEST AND VERIFICATION. Upon the completion of each flight test, the recording medium shall be removed and analyzed to verify the proper recording of the specified data.

PART II COMMUNICATION SYSTEM
SECTION J FLIGHT DATA RECORDER

1.0 SCOPE

This section of this specification establishes the requirements for performance, design, test and qualification of equipment identified as the Flight-Data Recorder Subsystem as applied to the prototype airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The Flight-Data Recorder shall record flight parameter data and shall protect this recorded data through a crash environment to provide information useful to a crash investigation.

3.1.1.1 Performance Characteristics. The system shall perform in accordance with ARINC Characteristic 541 and FAA TSO-C51a as revised and issued on December 29, 1965.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Flight-Data Recorder Subsystem shall not exceed 0.67 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Flight-Data Recorder Subsystem shall not exceed a mean expenditure of 2.6 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Pa[illegible] .2.2.1 of Part I.

3.1.2.2.2 Service and Access. The Flight-Data Recorder Subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 General. Refer to the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.4.2 Recorder Unit. The recorder unit shall meet the environmental requirements as specified in FAA TSO-C51a as revised and issued on December 29, 1965.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITIONS.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 10.

3.2.1.2 Detailed Interface Definition. The system shall provide interface with appropriate ARINC and FAA ground systems. The power requirements shall not exceed 25 VA at 115-volt, 400 cps. The system shall record data obtained from the Air-Data System.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The weight of the subsystem shall not exceed 42.0 pounds. The recorder unit shall be installed in the aft cargo area. A time and event marker shall provide for manual input of specific time, date, and aircraft identification on the recording medium.

3.3.2 through 3.3.11 — See Part I of this specification.

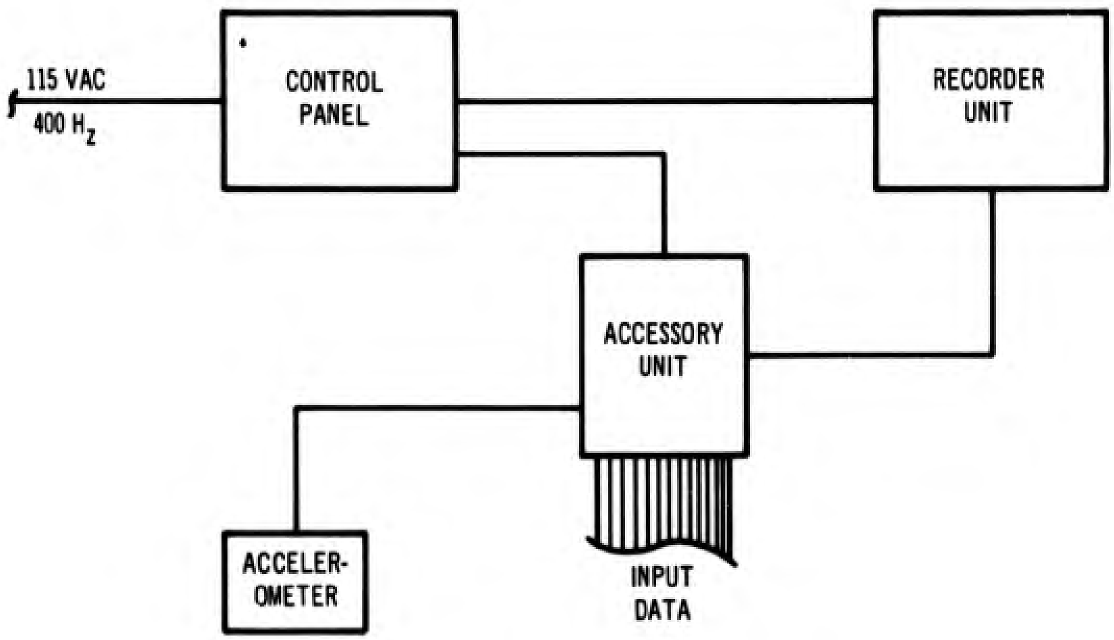


Figure 10. Flight Data Recorder

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Refer to Par. 4.3.4 of Part I.

4.4 RELIABILITY TEST AND ANALYSIS. Upon the completion of each flight test, the recording medium shall be removed and analyzed to verify the proper recording of the specified data.

PART III NAVIGATION SYSTEM
SECTION A LOW FREQUENCY (ADF)

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test, and qualification of equipment identified as the low-frequency Automatic Direction Finding (ADF) subsystem as applied to the prototype airplane. This subsystem shall provide bearing indication relative to a selected ground station.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE

3.1.1 Functional Characteristics. The completely redundant Automatic Direction Finding (ADF) systems shall operate in the frequency range of 190 to 1,750 kHz. The ADF systems shall provide the pilots with a back-up system to the Inertial Navigation System (INS) to obtain position fixes over routes not served by VHF NAV aids and provide a back-up means of position fixing during final approach over the LOM and LMM stations.

Each ADF system shall provide an audio input to the interphone system and bearing information to the HSI in accordance with ARINC Characteristic 550.

3.1.1.1 Performance Characteristics. The ADF system shall provide bearing and station identification signals at minimum ranges of 50 nmi from an ILS outer marker compass locator station (LOM) or 200 nmi from an RA of H-type low frequency radio station.

The two-sigma bearing error, including display error, shall not exceed ±5° at all relative bearings except 0° and 180° where the error shall not exceed ±3°.

The antenna installation shall not cause the ADF system to exhibit post over-the-station reversal. Over-the-station reversal shall be indicated by prompt and positive bearing pointer reversal.

The ADF equipment shall meet the performance requirements of RTCA Paper 158-61/Do-111, ARINC Characteristic 550, FAA TSO-C41b.

3.1.2 Operability. Refer to Par. 3.1.2. of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the ADF subsystem shall not exceed 0.02 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the ADF system shall not exceed a mean expenditure of 12.7 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The ADF subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The ADF antenna structure shall be designed to withstand, without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The ADF equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. All components which form a part of a functional man-machine interface shall be designed in accordance with human engineering principles outlined in Par. 3.1.2.5 of D6A10109-1, Flight Deck Subsystem Specification.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 11.

3.2.1.2 Detailed Interface Definition. The ADF system shall provide interface with appropriate ARINC and FAA ground systems.

Power requirements of each system shall not exceed 45 watts.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The maximum weight of each ADF receiver shall not exceed 21.2 pounds. Each receiver shall be capable of driving three ARINC loads. The detailed mechanical and electrical design of this equipment shall be subject to the requirements of FAA TSO-C41b, and ARINC Characteristic 550.

The ADF system consists of two receivers, two loop antennas, two sense antennas, two control panels, two sense-antenna couplers, and two quadrantal error correctors.

3.3.2 through 3.3.11 } See Part I of this specification.

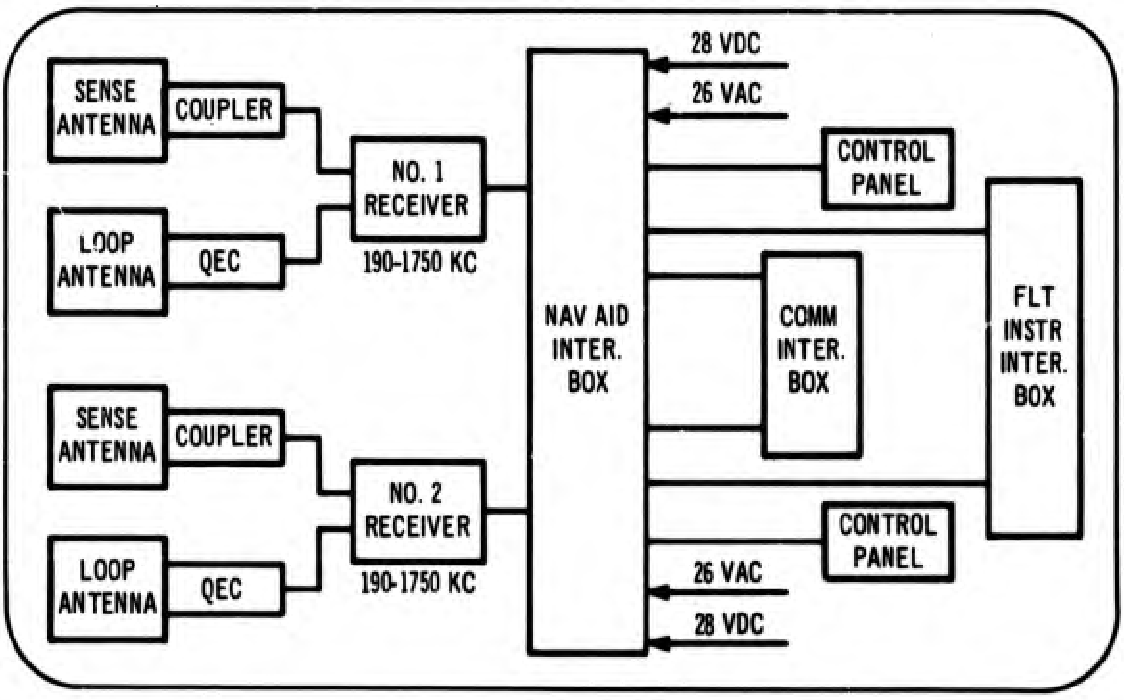


Figure 11. ADF No. 1 and No. 2

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

a. Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

b. The optimum antenna location shall be substantiated by testing on the 1/40th-scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO Specifications, or otherwise noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.3 Quadrantal Error (QE) Tests. The gross QE of the airplane shall be determined during a ground swing of the airplane on a surveyed site similar to a compass rose.

4.3.4.4 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements as specified in Par. 3.0 of this section.

4.3.4.4.1 Tune in an NDB or H-type facility in the 190- to 400-kHz frequency range during subsonic and supersonic flight, and at a minimum distance of 100 nmi. Select a VOR course which passes through the LF station and while flying ON COURSE by centering the HSI course deviation bar verify that the ADF No. 1 and No. 2 needles on both the Captain's and First Officer's HSI instruments agree with the selected course within ±3°.

4.3.4.4.2 Determine that the distance at which an NDB- or H-type station becomes unidentifiable exceeds 200 nmi when in ANTENNA MODE and when flying at both subsonic and supersonic speeds at altitudes of 25,000 feet and above.

4.3.4.4.3 Verify that quadrantal error does not exceed ±3.0 on the 0° and 180° relative bearings or ±5° on any other relative bearing.

4.3.4.4.4 Verify that sense antenna location is satisfactory as indicated by the absence of post-over-the-station needle reversal.

4.4 RELIABILITY TEXT AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION B-VHF NAVIGATION (VOR/LOC)

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test, and qualification of equipment identified as VHF Radio Navigation (VOR/LOC) Subsystem as applied to the prototype airplane. This subsystem shall provide guidance information relative to a selected ground station operating in the frequency range of 108.00 to 117.95 mHz. The VHF radio navigation receiver shall provide outputs for deviation, flag-alarm movements, bearing, to-from indication and audio.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. Dual VHF navigation receiving systems shall provide enroute navigation information from VHF navigation aids operating in the frequency range of 112.0 to 117.95 mHz when operating in the VOR mode, and terminal navigation information when operating in the LOC mode. The localizer frequency coverage shall be 108 to 112 mHz. When operating in the LOC mode, the VHF navigation system shall provide localizer path displacement information in accordance with FAA operational performance Category III ILS requirements.

Each VHF navigation receiver shall provide outputs for deviation indicators, bearing indicators, to-from indicator, autopilot, flag-alarm movements, and audio, in accordance with ARINC Characteristics 547.

3.1.1.1 Performance Characteristics. The VHF navigation receiving system shall provide reliable bearing information and station identification at nominal cruise altitude at clear line-of-sight ranges of 0 to 200 nmi from a Category A, 200 watt, VOR station (as defined in Par. 3.3.4.2 of ICAO Annex 10). The two sigma errors including display errors shall not exceed ± 4.0°.

The LOC receiver shall provide lateral guidance at ranges of 25 nmi, compatible with Category III Instrument Landing System during the final leg of an approach and landing.

The VHF navigation system shall be capable of receiving VOR signals at all angles of azimuth and the displayed bearing information between the dual systems shall be within ±3.0°.

The LOC system shall not display a flag when flying at all headings within the full-scale course deviation instrument limits when flying a distance of less than 45 nmi from the localizer ground station.

The dual VHF navigation system shall be independent to the extent that a fault in one receiver or that system's transmission line will not degrade the receiving ability of the other system by more than 10 db.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the VOR/LOC subsystem shall not exceed 12.10 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the VOR/LOC subsystem shall not exceed a mean expenditure of 24.9 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The VOR/LOC subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The VHF navigation antenna shall be designed to withstand, without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Antenna materials shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.4.2 The VHF navigation equipment, except the antenna shall be designed to withstand, without damage or impairment of performance, the pressurized environmental condition of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 12.

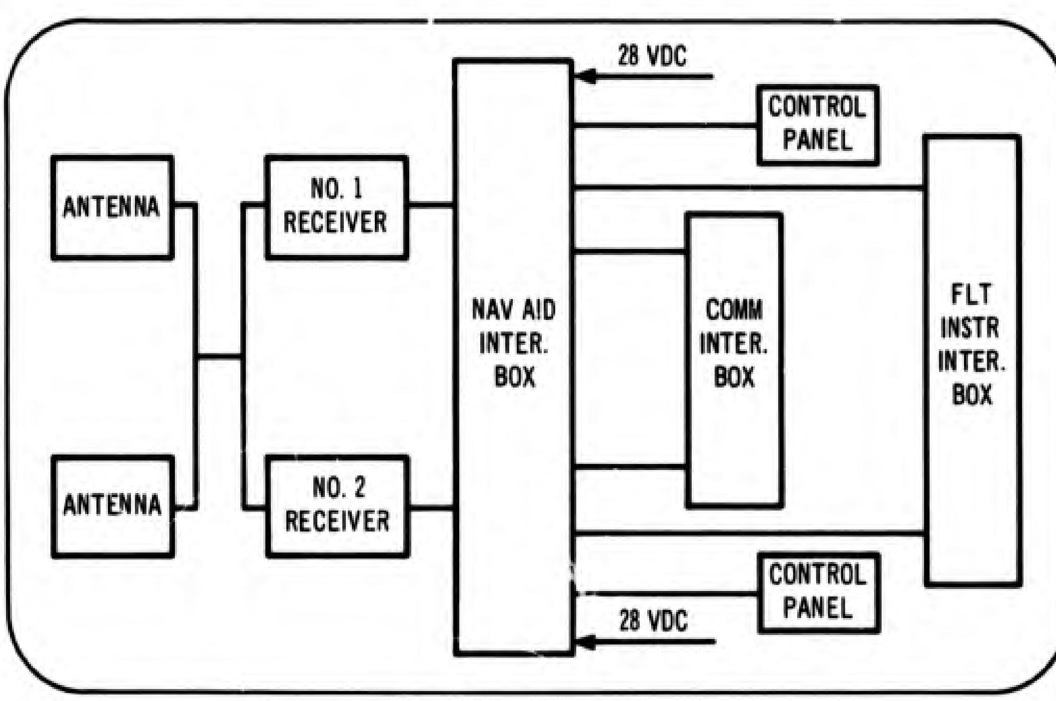


Figure 12. VOR No. 1 and No. 2

3.2.1.2 Detailed Interface Definition. The system shall provide interface with appropriate ARINC and FAA ground systems. The power requirements of the VHF navigation system shall not exceed 27.5 volts direct current, 45 watts.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The VHF navigation equipment shall be inherently stable in mechanical construction and electrical characteristics, and shall be suitable for extended commercial aircraft use. The detailed mechanical and electrical design shall be subject to the requirements of FAA TSO-C40b, except as noted herein.

Built-in self-test, continuous self-monitoring and self-calibrating capability shall be provided to detect and isolate system malfunctions.

A capability for remote control and readout of this function shall be provided.

The maximum weight of the VHF navigation unit shall not exceed 123.7 pounds.

The system shall consist of units per ARINC 410 and 547.

High reliability shall be assured either through constant integrity monitoring and self-calibration techniques and/or redundancy of major components coupled with voting logic.

A VHF navigation receiving system shall consist of an omnidirectional, horizontally polarized, antenna for the VOR/LOC functions, the antenna being common to both systems; the necessary coaxial transmission lines; a VHF navigation receiver; a control panel located in the flight deck area with provisions for frequency and mode selection and audio-level adjustment; and a course deviation indicator installed in the flight instrument panel.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

a. Antenna Radiation Pattern.

The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements as specified in Par. 3.0 of this section.

4.3.4.3.1 VHF NAV (VOR/LOC) System

4.3.4.3.1.1 Flight tests of the VHF NAV (VOR/LOC) systems in the Model B-2707 airplane will be conducted in both subsonic and supersonic regimes. In either case, VOR signals will be received over line-of-sight paths.

4.3.4.3.1.2 During flight at subsonic speeds and at altitudes up to 18,000 feet MSL, tune NAV No. 1 and NAV No. 2 receivers to the same "Low Altitude" VOR facility. Adjust the VOR/ADF Controls on both the Captain's and First Officer's HSI instruments to VOR, and on the Flight Mode Selector Panel, adjust both VOR/LOC Course Set Controls for the desired VOR bearing. The bearing needles for No. 1 and No. 2 VOR shall indicate the selected bearing and shall agree within ±3.0°. The VOR/LOC Course Deviation Bar will align with the Course Cursor when the airplane is flying the selected course toward the VOR facility and the "TO" flags will be displayed on both HSI instruments. As the airplane passes over the facility, the "TO" flags shall disappear and the "FROM" flags shall appear on both HSI's.

4.3.4.3.1.3 The conditions called out in Par. 4.3.4.3.1 apply for flight above 18,000 feet MSL except that "High Altitude" VOR Facilities will be utilized.

4.3.4.3.1.4 As each succeeding High Altitude VOR is selected and flown, monitor the channel aurally for possible cochannel interference arising from the altitude at which the flight is conducted.

4.3.4.3.1.5 Note whether popping flags are in evidence indicating weak signal conditions.

4.3.4.3.2 ILS Localizer Flight Test.

4.3.4.3.2.1 With the airplane at a distance of approximately 45 nmi from the ground station, fly the airplane to intercept the runway localizer course first from one side, and then the other, Observe that:

a. The LOC warning flag is hidden.

b. The course deviation needles on both HSI's present the correct information to enable the pilot to center the airplane on the course center line.

4.3.4.3.2.2 At a distance of 20 nmi from the localizer ground station, approximately on course center line and at an altitude of 6,000 feet, fly a 360° circle either to right or left maintaining a 10° bank angle. Observe that a usable localizer signal is obtained at all headings within the localizer antenna design limit as indicated by the localizer flag remaining hidden.

4.3.4.3.2.3 While on an inbound approach and on localizer course centerline, maneuver the airplane in normal pitch attitudes using the glide slope (G/S) deviation needle to define the upper and lower limits of the glide path. Observe that the pitch attitude of the airplane does not affect the localizer course deviation bar.

4.3.4.3.2.4 Flying inbound and with the airplane centered on both LOC and G/S paths engage the autopilot and observe that:

a. There is no mutual interference between the localizer and the autopilot.

b. The autopilot maintains the airplane on the localizer course.

4.3.4.3.2.5 Flying inbound with the airplane centered on both LOC and G/S paths, engage the flight director and note that:

a. There is no mutual interference between the localizer and the flight director.

b. The flight director information supplied to the pilot allows him to maintain the airplane on the localizer course.

4.4 RELIABILITY TEXT AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION C GLIDE SLOPE

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test and qualification of equipment identified as the Glide Slope Subsystem as applied to the prototype airplane. This subsystem shall provide glide path information relative to a selected ground station operating in the frequency range of 329.3 to 335.0 mHz. The glide slope receiver shall provide outputs for deviation, and flag-alarm movements.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The glide slope receiving subsystem shall provide terminal glide path information from VHF navigation aids operating in the frequency range of 329.3 to 335.0 mHz in accordance with FAA operational performance Category III ILS requirements.

The glide slope receiver shall provide outputs for deviation indicators and flag-alarm movements in accordance with ARINC Characteristics 551.

3.1.1.1 Performance Characteristics. The glide slope equipment shall meet the requirements of ARINC Characteristic 551, RTCA Document DO-132 and FAA TSO-C34b. The Glide Slope receiver shall provide glide path deviation at ranges of 15 nmi, compatible with Category III Instrument Landing System during the final leg of an approach and landing.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Glide Slope Subsystem shall not exceed 1.80 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Glide Slope Subsystem shall not exceed a mean expenditure of 16.7 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycles. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The Glide Slope Subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The glide slope antenna shall be designed to withstand, without damage or impairment of performance the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Antenna materials shall be compatible with the hydraulic fluid tube used in the supersonic transport airplane.

3.1.2.4.2 The glide slope equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 13.

3.2.1.2 Detailed Interface Definition. The system shall provide interface with appropriate ARINC and FAA ground systems.

The power requirement of the unit shall not exceed 27.5 volts direct current, 15 watts.

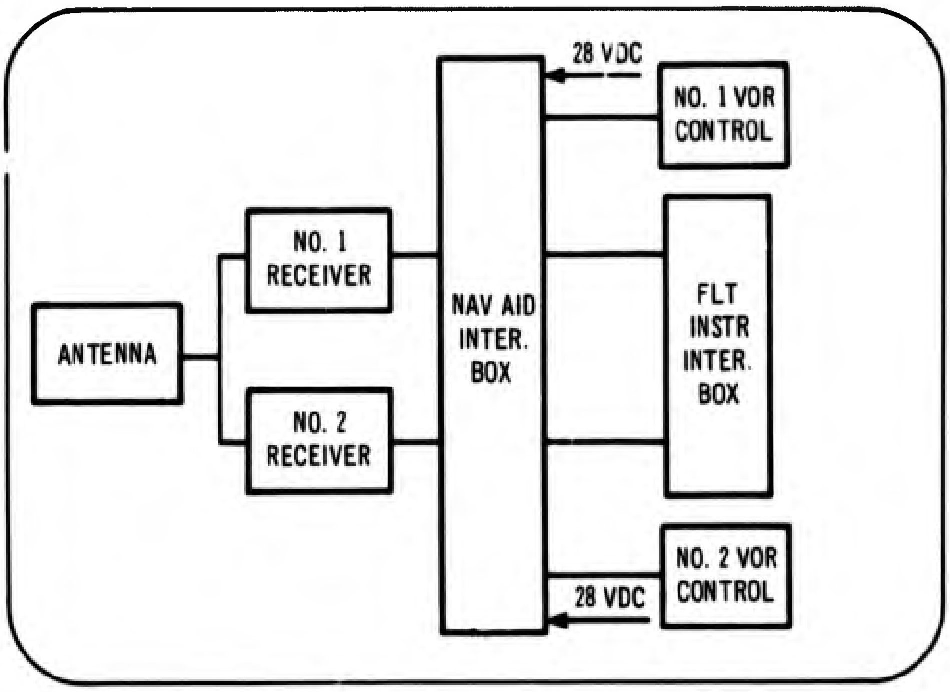


Figure 13. Glide Slope, No. 1 and No. 2

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The glide slope equipment shall be inherently stable in mechanical construction and electrical characteristics, and shall be suitable for extended commercial aircraft use. The detailed mechanical and electrical design shall be subject to the requirements of FAA-TSO-C34b, and ARINC Characteristic Numbers 410 and 551, except as noted herein.

Built-in self-test, continuous self-monitoring and self-calibrating capability shall be provided to detect and isolate system malfunctions.

A capability for remote control and readout of this function shall be provided. High reliability shall be assured either through constant integrity monitoring and self-calibration techniques and/or redundancy of major components coupled with voting rights.

A Glide Slope receiving subsystem shall consist of a horizontally polarized antenna, the necessary coaxial transmission lines, a Glide Slope receiver, and glide path deviation indication installed in the flight instrument panel.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATIONS. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

a. Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements as specified in Par. 3.0 of this section.

4.3.4.3.1 With the airplane flying inbound on the LOC course and at a distance approximately 25 nmi from the glide slope transmitter, intercept the glide slope course. Observe that a usable signal is received at a distance not less than 10 nmi as indicated by glide slope warning flag retraction.

4.3.4.3.2 With the airplane on course (both localizer and Glide Slope) turn the airplane 30° right from LOC on course until full-scale deviation of the HSI course deviation needle is observed. Note that the glide slope flag remains hidden. Make a 30° left turn until full-scale deviation of the course needle is observed. The glide slope flag shall remain hidden as before. (The glide slope flag is not required to appear for such deviations from the localizer center line as the G/S signal is usable beyond these limits).

4.3.4.3.3 While making a normal approach ON PATH, engage the autopilot and note that:

a. There is no mutual interference between the Glide Slope system and the autopilot.

b. The autopilot maintains the airplane on the glide path.

4.3.4.3.4 With the airplane inbound and "ON PATH" engage the flight director and note that:

a. There is no mutual interference between the Glide Slope system and the flight director.

b. The flight director information supplied to the pilot allows him to maintain the airplane on the glide path.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION D MARKER BEACON

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, tests and qualification of equipment identified as the Marker Beacon Subsystem as applied to the prototype airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The Marker Beacon equipment shall provide visual and aural indication of passage over an enroute or terminal marker station. It shall be designed to receive Marker Beacon signals on a single fixed frequency of 75 mHz. The equipment shall provide an overall frequency stability not exceeding ±0.005 percent of the channel frequency under service conditions.

3.1.1.1 Performance Characteristics. The Marker Beacon shall meet the performance requirements of RTCA Paper 87-54/DO-57A and FAA TSO-C35c.

The Marker Beacon system shall provide a white light indication, a 3,000-Hz audio tone, and intelligible station identification when operated over an airways or terminal marker station. The Marker Beacon system shall provide an amber light indication, a 1,300-Hz audio tone and intelligible station identification when operated over a middle marker transponder. The Marker Beacon system shall provide a blue light indication, a 400-Hz audio tone, and intelligible station identification when operated over an outer marker transponder.

The Marker Beacon antenna shall provide a downward directed polarized pattern so that the electric vector is parallel to the airplane fuselage.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Marker Beacon subsystem shall not exceed 0.01 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Marker Beacon subsystem shall not exceed a mean expenditure of 1.9 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The Marker Beacon subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environmental.

3.1.2.4.1 The Marker Beacon antenna shall be designed to withstand without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperatures. Minus 50°F to plus 420°F.

b. Temperature Shock. Minus 50°F to plus 420°F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fuel Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The Marker Beacon equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 14.

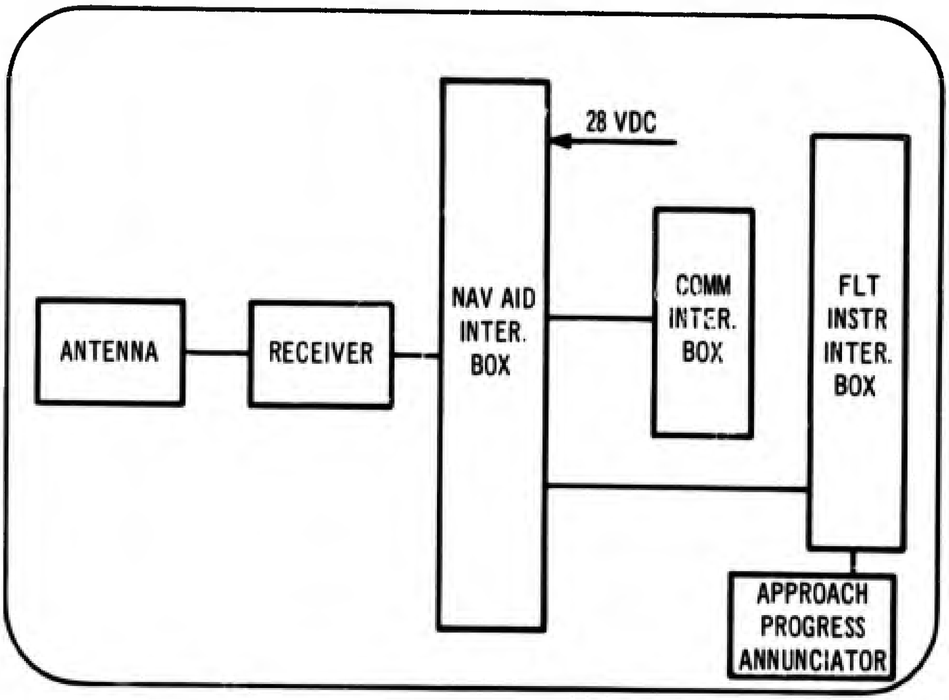


Figure 14. Marker Beacon

3.2.1.2 Detailed Interface Definition. The system shall interface with appropriate ARINC and FAA ground systems.

The power requirements of the system shall not exceed 20 watts and 27.5-volts direct current.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The Marker Beacon equipment shall be inherently stable in construction and electrical characteristics. The detailed mechanical and electrical design shall be per requirements of FAA TSO-C35c. The weight of the system shall not exceed 12.0 pounds.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility with the following exceptions:

a. Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a free-space antenna range.

The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Documentation. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements specified in Par. 3.0 of this section.

4.3.4.3.1 Pass directly over a fan-type marker at 5,000-feet altitude. The airway marker light (white) shall glow or flash regularly for a minimum distance of one mile. With the pilots' interphone set for marker beacon reception, a steady or coded 3000-Hz signal shall be heard in the headset.

4.3.4.3.2 Repeat at 25,000 feet.

4.3.4.3.3 During letdown on the glide path, note that the autopilot glide slope extension function as satisfied as the middle marker antenna pattern in intercepted.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION E DISTANCE MEASURING EQUIPMENT (DME)

1.0 SCOPE

This section of the specification establishes the requirements for performance, design, test and qualification of equipment identified as distance measuring equipment (DME) subsystem as applied to the prototype airplane. This subsystem, shall provide an accurate identification of slant-range distance from the aircraft to a selected DME ground facility.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. Each DME system shall provide outputs, as specified in ARINC Characteristic 521D, for the flight-director computer, the flight-deck indicators, and for the suppression of other L-Band equipment.

3.1.1.1 Performance Characteristics. The DME systems will transmit in the frequency range of 1,025 to 1,150 mHz, and shall operate on 126 channels with full provisions for expanding operation to 252 channels. Each DME system shall be able to lock-on and track a standard VOR/TAC ground facility at ranges from 0 to 1,800 knots with an accuracy not exceeding ±0.7 nmi.

Each DME system shall be capable of obtaining continuous replies from the ground transponder and displaying range information at all azimuth angles and at bank angles up to 10°.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the DME subsystem shall not exceed 0.02 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the DME subsystem shall not exceed a mean expenditure of 2.6 direct maintenance manhours per

1,000 flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The DME subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The DME antenna shall be designed to withstand without damage or impairment of performance the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50° F to plus 420° F.

b. Temperature Shock. Minus 50° F to plus 420° F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The distance measuring equipment, except the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 15.

3.2.1.2 Detailed Interface Definition. The DME system shall provide interface with appropriate ARINC and FAA ground systems.

The power requirement for the system shall not exceed 137-watts alernating current.

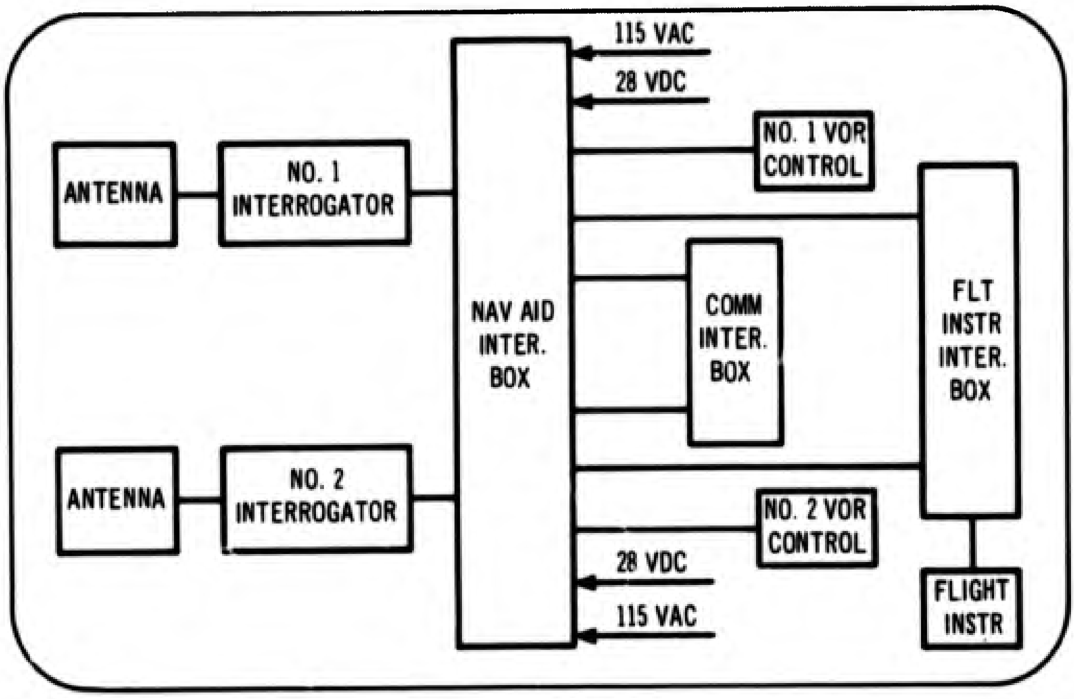


Figure 15. DME, No. 1 and No. 2

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The DME system shall be inherently stable in mechanical construction and electrical characteristics, and shall be suitable for extended commercial aircraft use. The detailed mechanical and electrical design shall be subject to the requirements of FAA TSO-C66.

The DME antenna shall be vertically polarized when mounted on the top or bottom of the airplane. A search range limit over-ride switch shall be provided on the control panel.

The weight of the DME shall not exceed 91.3 pounds.

The DME systems shall incorporate a system confidence check activated by a push-to-test switch located convenient to the pilot.

A dual DME system shall consist of two airborne interrogators, two antennas, two flight deck control units and two flight deck indicators.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility, with the following exceptions:

The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range.

The optimum antenna location shall be substantiated by testing on the same scale model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 Formal Qualification Tests. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA - TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements specified in Par. 3.0 of this section.

4.3.4.3.1 The DME System's operational capabilities shall be demonstrated during enroute and terminal area maneuvers. The following outlines the tests deemed necessary to demonstrate the specified performance.

4.3.4.3.2 Beginning at a distance of 5 to 10 nmi from a DME facility, fly inbound and pass over the DME station at an altitude of 1,000 feet. Continue outbound and at a distance of 5 to 10 nmi outbound from the station, initiate a maximum rate of climb to an altitude of 30,000 feet or the airplane maximum-certified altitude.

4.3.4.3.3 Fly outbound maintaining a relative heading from to the ground station at 180° at an altitude of 30,000 feet or the airplane maximum-certified altitude to a distance of at least 160 nmi from the ground station.

4.3.4.3.4 At an altitude of 30,000 feet fly the airplane through two 360° turns, one left and one right, at bank angles of not more than 10° with the center of the circle 160 nmi from the ground station.

4.3.4.3.5 Repeat the flight profile of Par. 4.3.4.7 at a distance of 60 nmi from the DME ground station at 50,000 feet.

4.3.4.3.6 At an altitude of 50,000 feet or above, fly the airplane inbound and pass over the DME facility.

4.3.4.3.7 Perform an inbound letdown from 30,000 feet or above at the airplane normal maximum rate-of-descent to arrive at an altitude of 5,000 feet 5 nmi from the DME station.

4.3.4.3.8 At an altitude of 5,000 feet fly a 90° left or right sector of an orbit around a DME ground station on maintaining a distance of 5 nmi from the station.

4.3.4.3.9 Determine that the DME systems does not interfere with any other airplane system and is not interfered with by any other system.

4.3.4.3.10 Cycle any airplane appendages, such as flaps, landing gear, etc., through their full range of travel and check for possible interference with the DME operation.

4.3.4.3.11 Actuate the "push-to-test" function on one DME system and check for adverse affects on the other DME system. Repeat, turning one DME system off and on.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION F RADIO ALTIMETER

1.0 SCOPE

This section of the specification establishes requirements for performance, design test and qualification of equipment identified as Low-Range Radio Altimeter Subsystem as applied to the prototype airplane. This subsystem shall provide a continuous and accurate indication of the aircraft height above the terrain for the approach or landing phases of flight operation.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The radio altimeter system shall provide a continuous and accurate indication of airplane height above the terrain during all normal airplane approach altitudes from 2,500 feet to touch down in all weather conditions.

3.1.1.1 Performance Characteristics. The radio altimeter shall provide reliable altitude information over all type of terrain from 0 feet at touchdown to 2,500 feet, with a displayed accuracy of ± 5 feet or ± 5 percent whichever is greater. The completely redundant Low-Range Radio Altimeter System shall operate at a carrier frequency of 4.3 GHz.

The radio altimeter shall provide the previously stated displayed accuracy under the following conditions:

a. Pitch angle of 0 ± 5° about the airplane mean approach altitude.
b. Roll angle of 0 ± 20°.
c. Forward velocity from minimum approach speed to 200 knots.

The radio altimeter, over level ground, shall track the actual altitude of the airplane without erratic or oscillatory altitude indication.

At an altitude of 200 feet or less, encountering a terrain change of not more than 10 percent of the airplane altitude shall not cause the radio altimeter to unlock, and the displayed response to such a condition shall not exceed 0.1 sec. At higher conditions of terrain changes, in the event of altimeter lock, the altimeter shall require the signal within one second.

The radio altimeter, when exercised in the self-test or push-to-test operation, shall be tested at a simulated altitude of less than 500 feet.

The radio altimeter shall provide to the flight crew a positive failure warning display when a system malfunction occurs, a loss of power, the absence of a ground return within the specified range of operating altitudes, or when the airplane exceeds the maximum operating altitude of the radio altimeter.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Low-Range Radio Altimeter Subsystem shall not exceed 9.85 per 100,000 schedule flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal deviations are shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours shall not exceed a mean expenditure of 44.2 direct-maintenance manhours per 1,000 flight hours on and off the airplane, based upon an average flight length of 1.75 hours.

3.1.2.2.2 Service and Access. The Low-Range Radio Altimeter Subsystem shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of system components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 The Low-Range Radio Altimeter antenna shall be designed to withstand, without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperatures. Minus 50° F to plus 420° F.

b. Temperature Shock. Minus 50° F to plus 420° F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fuel Compatibility. Antenna materials shall be compatible with the hydraulic fluid to be used in the supersonic transport airplane.

3.1.2.4.2 The Low-Range Radio Altimeter equipment, with the exception of the antenna, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 16

3.2.1.2 Detailed Interface Definition. The Low-Range Radio Altimeter Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The power requirements for the system shall not exceed 115-volts ac, 80 VA.

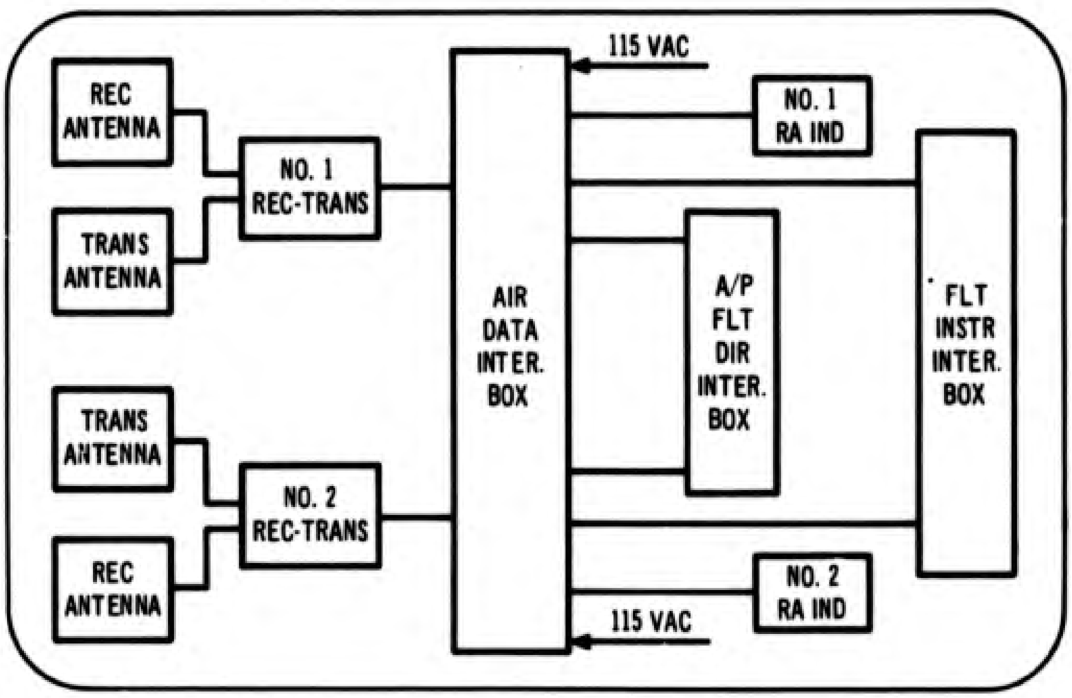


Figure 16. Radio Altimeter No. 1 and No. 2

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The Low-Range Radio Altimeter subsystem shall consist of units per ARINC 552. The weight of the transceiver shall not exceed 47.2 pounds. Completely redundant systems will be provided.

The antenna shall be designed for flush mounting and shall be installed to have an unobstructed view cone of 120° maintained between antenna locations and aircraft objects.

3.3.2 through 3.3.11 } See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.1.1 Antennas. All antenna development testing shall be conducted at the vendor facility, with the following exception:

a. Antenna Radiation Pattern. The antenna radiation pattern data shall be obtained using the 1/40th-scale model of the SST on a freespace antenna range. The optimum antenna location shall be substantiated by testing on the same model.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, and selected design and performance requirements specified in the applicable ARINC Characteristics and FAA-TSO specifications, or otherwise as noted herein.

4.3.4.1 Subsystem Integration Tests. Subsystem integration testing shall be conducted to verify interface and interference requirements as specified in Par. 4.3.4 of Part I, or as otherwise noted herein.

4.3.4.2 Airplane Ground Tests. Airplane ground testing shall be conducted to verify system installation performance in the airplane environment as specified in Par. 4.3.4 of Part I.

4.3.4.3 Airplane Flight Test. This paragraph delineates the airplane flight test demonstrations which verify the design and performance requirements specified in Par. 3.0 of this section.

4.3.4.3.1 The radio altimeter system operational capabilities shall be demonstrated while flying terminal area maneuvers such as approaches, fly-bys, and landings in both the coupled (automatic) and manual (Flight Director) modes. The following outlines the tests deemed necessary to demonstrate the specified performance of Par. 3.0 of this section.

4.3.4.3.2 While flying at an altitude below the maximum readout altitude of the altimeter over flat terrain or water, observe the altimeter operation during the following flight operations and maneuvers. Verify that there are no detrimental effects:

a. As the flaps, landing gear, and other airplane appendages, that may interfere with the altimeter antenna patterns, are cycled through their full range of travel

b. As the airplane is banked right and left until either the altimeter flag shows or the bank angle is beyond that expected for normal flight operation

c. As one system is turned off and on

4.3.4.3.3 With the equipment using the operating altimeter system, observe that the operation of the altimeter and equipment during an approach and landing are performing their intended function

4.3.4.3.4 Observe that there are no mutual interference effects between the altimeter systems and any other system on the airplane during the flight.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION G WEATHER AND MAPPING RADAR

1.0 SCOPE

This part of this specification establishes the requirements for performance, design, tests, and qualification of one type of equipment identified as the Weather and Ground Mapping Radar subsystem as applied to the prototype airplane. This subsystem shall provide flight crew with visual intelligence indicating the presence of precipitating clouds and their location with respect to the aircraft.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The Weather and Ground Mapping Radar system shall provide the flight crew with visual information indicating the presence of and, to a relative degree, the intensity of severe weather (thunderstorm) activity. The radar system shall also provide a visual radar map of the surrounding surface area.

3.1.1.1 Performance Characteristics.

3.1.1.1.1 General. The Weather and Ground Mapping Radar system shall conform to FAA TSO-C63A except Pars. 1.5, 1.6, 2.4, 2.14 and 3.0.

3.1.1.1.2 Normal Weather Mapping (detection). The system shall display the location (range and azimuth) of severe thunderstorms within the scanned area along the flight path during the aircraft climb and descent conditions. The system, shall meet or exceed the range performance requirements criteria described by ARINC Characteristic 564 for X-band radar equipment. The system shall detect and display the location (range and azimuth) of severe thunderstorm activity with tops exceeding 40,000-feet altitude within the scanned area, and a minimum range of 250 nmi along the flight path (during the cruise phase of the aircraft flight profile).

3.1.1.1.3 Contour Weather Mapping (iso-echo). The system shall provide a direct video display of the thunderstorm cores and their relative positions. The system shall detect the preselected signal level and employ adequate blanking techniques to the video to display a minimum of two echo strengths.

3.1.1.1.4 Ground Mapping. The system shall provide a radar display map of the surrounding surface area which can be used by the flight crew for identification of major landmarks at cruise altitude and at a minimum range of 180 nmi.

3.1.1.1.5 Performance Monitoring (self-test). The system shall provide a self-test feature as an integral part of the system design. The self-test mode shall provide control and display for confidence testing from the flight deck of the critical components and circuitry.

3.1.1.1.6 Controls and Display. The system shall provide adequate controls to the flight crew for performing the function of power on/off, mode selection, antenna control and adjusting the display.

The display shall include the following characteristics: the display section shall be a minimum of ±90° azimuth and have ranges commensurate with the requirements of ARINC Characteristic 564. Brightness - the display shall have a Bright tube.

3.1.1.1.7 Antenna Stabilization. The system shall include provisions for correcting the antenna to the reference attitude during maneuvers of the aircraft. The mechanical limits of antenna stabilization correction shall be a minimum of ±30° in elevation (pitch) and ±35° in roll.

3.1.1.1.8 Radome. The radome shall consist of an electromagnetically transparent and aerodynamically structural portion of the aircraft forward of the radar antenna mounting bulkhead.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Weather and Ground Mapping Radar subsystem shall not exceed 1.50 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration.. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Weather and Ground Mapping Radar subsystem shall not exceed a mean expenditure of 118.9 direct maintenance manhours per 1,000-flight hours on and off the airplane, not including servicing of consumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The system shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of systems components.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4. Environment.

3.1.2.4.1 Antenna and Radome. The antenna and radome shall perform in the additional environmental conditions as listed below:

a. Operating Temperature. Antenna - 50° to 400° F.
Radome - 50° to 450° with short periods to 500°F.

b. Temperature Shock. Antenna - 50° to 400° F.
Radome - 50° to 500° F.

c. Icing. The most severe icing conditions encountered in flight.

d. Hydraulic Fluid Compatibility. Radome materials shall be compatible with the hydraulic fluid to be used in the supersonic airplane.

e. Hail Impact. There shall be no extensive damage to the radome as a result of an encounter with 1.0-inch diameter hailstones at Mach 0.95 at an altitude of 40,000 feet.

f. Rain Erosion. Protection from rain erosion will be required for the radome.

3.1.2.4.2 Controls and Displays. The displays shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.5 Human Performance. Refer to Par. 3.1.2.5 of Part I.

3.1.2.6 Safety. Refer to Par. 3.1.2.6 of Part I.

3.2 SUBSYSTEM DEFINITION.

3.2.1 Interface Requirements. The Weather and Ground Mapping Radar system design shall provide for proper interface with the flight crew and the mechanical, electrical and condition environments of the aircraft.

3.2.1.1 Schematic Arrangement. See Fig. 17.

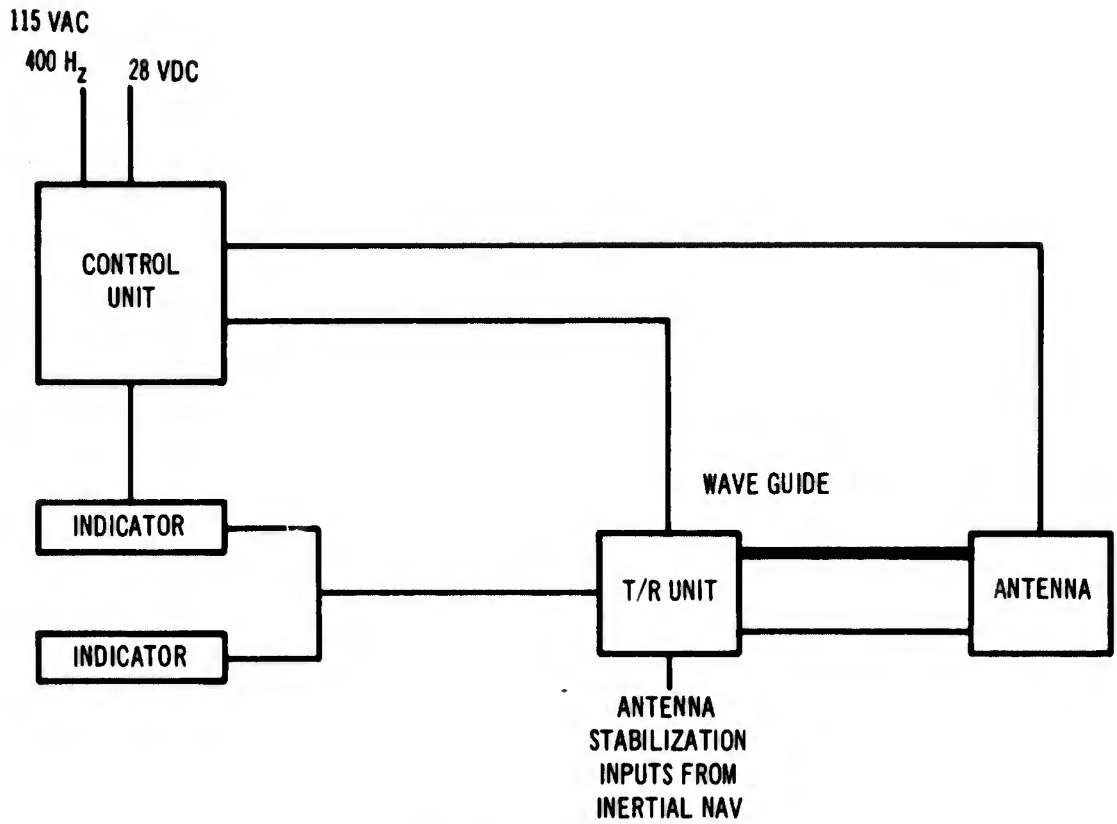


Figure 17. Weather Radar

3.2.1.2 Detailed Interface Definition. The Weather and Ground Mapping Radar system design shall provide for the following interfaces:

Flight Crew - Operation
Mechanical - Mounting and connecting provisions
Electrical
 Power - Primary electrical power inputs
 Inertial Navigation - Roll and pitch inputs
Environmental Control - Cooling air and pressurizations provisions.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The Weather and Ground Mapping Radar system shall operate in the X-band region of the microwave spectrum. The components (antenna, transmitter, receiver, and power supplies,) necessary for the transmit, receive and stabilization functions shall be remotely located within the nose of the aircraft. Primary electrical power and attitude reference signals for the remote-radar equipment shall be obtained from the aircraft electrical system and inertial navigator respectively. All remaining components (excluding interconnecting cabling) shall be located within the flight deck compartment accessible to both pilots. The system shall provide the displays and controls required for selecting and adjusting each of the operational modes which include weather map, weather contour (iso-echo) map, ground map, and performance monitoring (self-test). The radome shall be aerodynamically contoured to match the lines of the aircraft structure. The forward portion of the radome shall be covered with an erosion protecting coating. Lightning diverter strips shall be installed on the radome in a manner to allow repair and/or replacement. Access shall be provided to the antenna for inspection and maintenance.

3.3.2 through 3.3.11 — See Part I of this specification.

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TESTS AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALFICATION TESTS. Refer to Par. 4.2 of Part I.

4 3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, or otherwise as noted herein.

4.3.4.1 Airplane Flight Tests. The flight testing of the radar/radome system shall be conducted on a limited no-interference basis with the primary objectives of the aircraft. In-flight performance shall be evaluated by functionally checking the system in all modes of operation during the following conditions:

a. Subsonic climb
b. Supersonic climb
c. Supersonic cruise
d. Subsonic cruise
e. Subsonic descent
f. Supersonic descent

Visual inspection shall be performed on the radome and the antenna installation after completion of each flight and discrepancies recorded. After completion of each 100-hour flight test period, the moisture content within the radome material shall be determined.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

PART III NAVIGATION SYSTEM
SECTION H INERTIAL NAVIGATION

1.0 SCOPE

1.0 This section of the specification establishes design, test, and qualification of the Inertial Navigation System as applied to the prototype airplane.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. The inertial navigation systems shall perform the enroute navigation function and shall provide: (1) geographical position of the aircraft on a world-wide basis independent of external aids (2) guidance information for manual or autopilot control of the aircraft along a predetermined course, and (3) aircraft attitude information for display, autopilot flight director operation, weather radar antenna stabilization, and other system functions.

3.1.1.1 Performance Characteristics. The system shall conform to the applicable portions of the Inertial Navigation System Technical Standard Order (TSO) and the ARINC Specification 561.

3.1.2 Operability. Refer to Par. 3.1.2 of Part I.

3.1.2.1 Reliability. Flight turnbacks or deviations resulting from malfunction of the Inertial Navigation Subsystem shall not exceed 11.85 per 100,000 scheduled flights. For reliability purposes, the term flight is interpreted to mean a nominal SST supersonic flight of 1.75-hour duration. Normal maintenance of the system is assumed. Definition of flight turnbacks or deviations is shown in Par. 3.1.2.1 of D6A10107-1.

3.1.2.2 Maintainability. Maintenance manhours for the Inertial Navigation Subsystem shall not exceed a mean expenditure of 179.4 direct maintenance manhours per 1,000 flight hours on and off the airplane, not including servicing of ccnsumables, based upon an average flight length of 1.75 hours.

3.1.2.2.1 Maintenance and Repair Cycle. Refer to Par. 3.1.2.2.1 of Part I.

3.1.2.2.2 Service and Access. The system shall not require servicing. Maintenance will be limited to unscheduled removal and replacement of systems.

3.1.2.3 Useful Life. Refer to Par. 3.1.2.3 of Part I.

3.1.2.4 Environment.

3.1.2.4.1 General. The Inertial Navigation Subsystem, except the Magnetic Azimuth Detector, shall be designed to withstand, without damage or impairment of performance, the pressurized environmental conditions of Par. 3.1.2.4 of Part I.

3.1.2.4.2 Detector. The Magnetic Azimuth Detector shall be designed to withstand, without damage or impairment of performance, the unpressurized environmental conditions of Par. 3.1.2.4 of Part I, except as modified below:

a. Operating Temperature. Minus 50°F to plus 450°F.
b. Temperature Shock. Minus 50°F to 450°F.

3.2 SUBSYSTEM DEFINITIONS.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Fig. 18.

3.2.1.2 Detailed Interface Definition. The Inertial Navigation Subsystem shall provide interface with appropriate ARINC and FAA ground systems.

The power requirements of the system shall not exceed 350 watts when running.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features. The weight of the system shall not exceed 300 pounds.

The Inertial Navigation system configuration shall consist of three ARINC 561 inertial navigation systems, dual magnetic azimuth detectors, dual converter units, a switching unit, and an interface unit. Each ARINC 561 Inertial Navigation system shall consist of a navigation unit, battery, control panel, and a display panel.

The switching unit shall contain the circuitry necessary to switch either the Number 1 system's electrical signal outputs (Captain's) or the Number 2 system's electrical signal outputs (First Officer's) with those from the Number 3 system, upon command.

The interface unit shall provide the means of interconnecting the components of the inertial navigation systems and of integrating the signals to and from other aircraft subsystems. The interface unit shall not contain any active devices.

3.3.2 through 3.3.11 — See Part I of this specification.

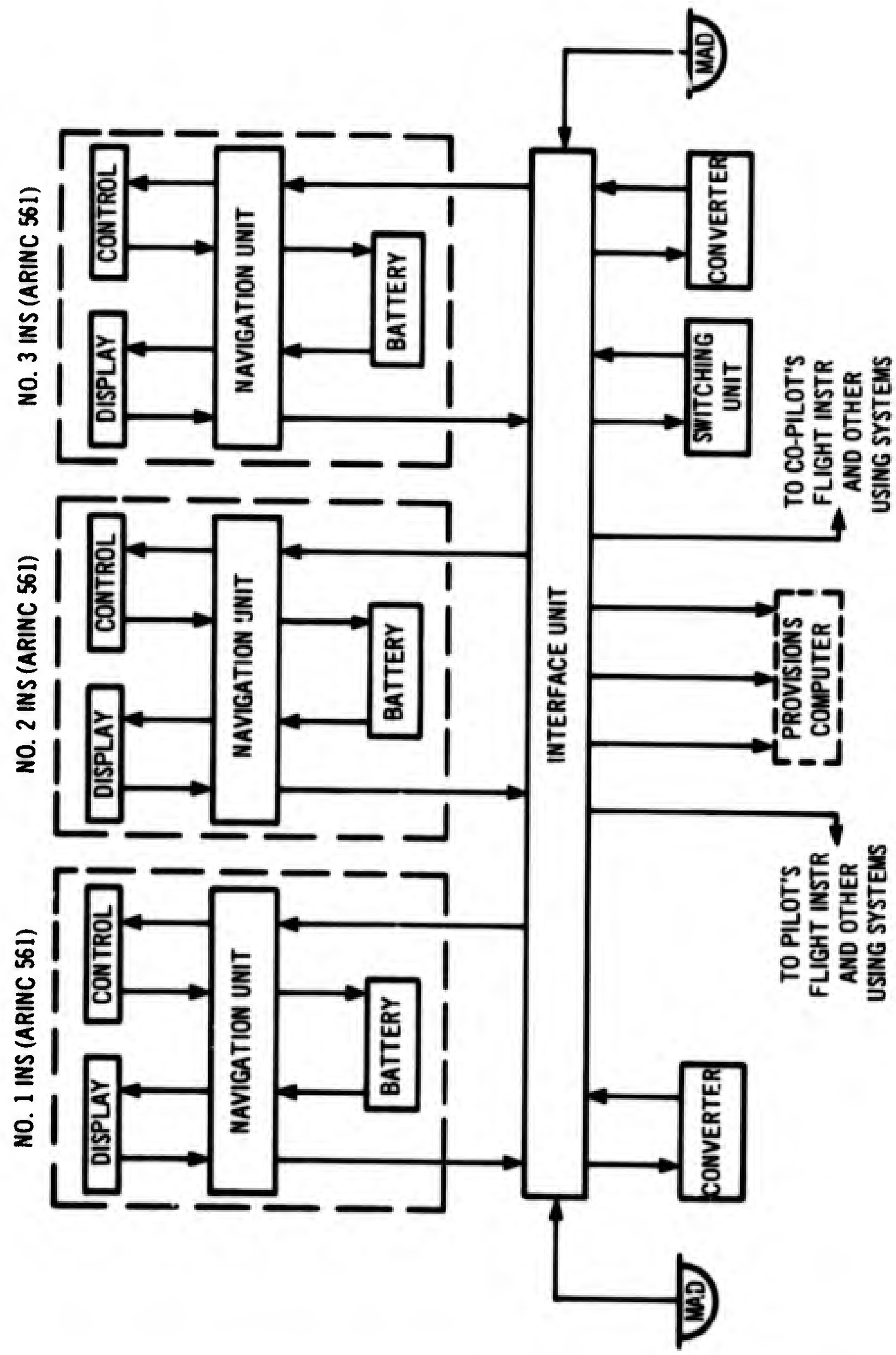


Figure 18. Inertial Navigation System

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TEST AND EVALUATION. Refer to Par. 4.1 of Part I.

4.2 PRELIMINARY QUALIFICATION TEST. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspections. Refer to Par. 4.3.1 of Part I.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstrations. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Subsystem bench tests shall be conducted to verify the equipment functions are in agreement with Par. 4.3.4 of Part I, or otherwise as noted herein.

4.3.4.1 Airplane Flight Tests. A simulated navigation run will be conducted to demonstrate subsystem accuracies after installation.

Subsystem performance in flight will be demonstrated by functionally checking the systems in appropriate modes of operation during the following conditions:

a. Supersonic cruise
b. Subsonic cruise
c. During maneuvers
d. Long acceleration phases
e. Ascents
f. Descents

After termination of flight, true-aircraft position and computed-aircraft position will be recorded.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.

1.0 SCOPE

This section of the specification establishes the requirement for performance, design, test and qualification of one type of equipment identified as the instrument displays and sensors as applied to the prototype airplane. This system shall provide the sources and computing of air data and the display of air and flight data.

2.0 APPLICABLE DOCUMENTS

See Sec. 2.0 of Part I.

3.0 REQUIREMENTS

3.1 PERFORMANCE.

3.1.1 Functional Characteristics. Instruments shall be provided to display parameters and derived functions of altitude, speed, attitude, heading and navigation data to the flight crew for use in the operation of the aircraft throughout its flight regime. The sensors providing flight data for displays shall also provide data for the flight control systems, flight data recorder and air traffic-control transponder.

3.1.1.1 Performance Characteristics.

3.1.1.1.1 Air-Data Computer. The air-data computer shall process pressure and temperature data related to the air-mass referenced speed and altitude of the aircraft and shall provide outputs required for:

a. Monitoring and control of the speed and altitude of the aircraft.

b. Monitoring aircraft performance and operating limits.

c. Automatic air-to-ground altitude reporting.

d. Vertical navigation.

3.1.1.1.1.1 Inputs. The air-data computer shall accept pneumatic inputs of total pressure and indicated static pressure from the aircraft pitot-static system and electrical inputs of indicated total air temperature and wing-sweep angle. The range of the input functions shall be as follows:

Function	Range
Total pressure (Pt)	4.0 to 75.0 inches of Hg
Indicated static pressure (Psi)	0.8 to 31.1 inches of Hg
Indicated total air temperature (t_{ti})	-70°C to 325°C
Wing-sweep angle (Λ)	20° to 72°

3.1.1.1.1.2 Outputs. The air-data computer shall provide the following output parameters and functions of these parameters:

Function	Range
Pressure Altitude (h_p)	-1,000 to 80,000 feet
Calibrated Airspeed (V_c)	60 to 800 knots
Mach Number (M)	0.2 to 3.0 Mach
Total Air Temperature (t_t)	-70°C to 325°C
True Airspeed (V_t)	150 to 1,800 knots
Static Air Temperature (t_s)	-100°C to 60°C
Non-Standard Day Temperature (t_s)	± 45°C
Pressure Altitude Hold (h_p)	0 to ±1,000 feet
Pressure Altitude Rate (h_p)	0 ± 20,000 feet/minute
Mach Number Hold (M)	0 to ± 0.1 Mach
Calibrated Airspeed Rate ($\dot{V}_c$)	0 ± 300 knots/minute
*Maximum Operating Speed (V_{MO}/M_{MO})	See Fig. 18

*Maximum operating speed shall be defined by wing-sweep angle, pressure altitude, and a structural temperature limit.

3.1.1.1.1.3 Computer Calibration. The computer shall be calibrated in accordance with the equations and tabular data listed in NASA Technical Note D-822, "Tables of Airspeed, Altitude, and Mach Number."

3.1.1.1.1.4 Static Source Error Correction. All computer outputs shall be corrected for static source error.

3.1.1.1.1.5 Performance Monitoring and Failure Alarm Provisions. The computer shall contain performance monitoring and failure alarm provisions. As a minimum design requirement, the monitoring circuitry shall be capable of detecting loss of airplane electrical power, computer power supply failure, and electrical failures of the computing servos.

The computer shall supply signals suitable for actuating indicator flags and other warning devices. Warning signal outputs shall be provided for pressure altitude, calibrated airspeed, Mach number, total air temperature true airspeed, static air temperature, and non-standard day temperature functions. A detected failure of the pressure altitude output shall result in the disabling of the altitude reporting encoder.

3.1.1.1.1.6 Output Function Tolerances. The allowable error of the outputs of the computer shall be as listed in Table I. Intermediate range tolerances shall be defined by straight lines connecting the specified points. The listed tolerances shall include calibration errors, friction, hysteresis, repeatability, and test equipment errors. The tolerances listed in Table I shall be applicable over the ranges specified unless the test point falls outside the "performance envelope" shown in Fig. 19. Tolerances for test points which fall between the "performance envelope" and the "operating envelope" as shown in Fig. 19 shall be permitted to be increased by 50 percent over the values listed in Table I.

3.1.1.1.1.7 Frequency Response. The frequency response characteristics of the computer servos shall meet or exceed the following requirements:

a. Pressure Altitude

Altitude, ft	Input Amplitude, ft	Frequency, cps	Phase Lag, °
2,000	±25	0.25	10
65,000	±50	0.25	20

b. Calibrated Airspeed

Altitude, ft	Input Amplitude, knots	Frequency, cps	Phase Lag, knots
2,000	±2.0	0.25	10° at 150
50,000	±5.0	0.25	20° at 550

c. Mach Number

Altitude, ft	Input Amplitude, M	Frequency, cps	Phase Lag, M
35,000	±0.01	0.25	15° at 0.9
65,000	±0.02	0.25	20° at 2.7

3.1.1.1.1.8 Damping. The servos of the computer shall be damped so the overshoot in response to a step input shall not exceed 5 percent.

Table I. Output Function Tolerances

a.	Pressure Altitude
	± 20 feet from -1,000 to 1,000 feet
	± 25 feet at 6,000 feet
	± 24 feet at 12,500 feet
	± 140 feet at 70,000 feet
	± 300 feet at 80,000 feet
b.	Calibrated Airspeed
	± 5.0 knots at 60 knots
	± 2.0 knots at 100 knots
	± 2.0 knots at 200 knots
	± 5.0 knots at 800 knots
c.	Mach Number
	± 0.005 Mach at sea level
	± 0.010 Mach at 60,000 feet
	± 0.020 Mach at 80,000 feet
d.	Total Air Temperature
	± 0.75°C over range of -70°C to 270°C
e.	True Airspeed
	± 6.0 knots at 150 knots
	± 6.0 knots at 600 knots
	± 12.0 knots at 1,200 knots
	± 12.0 knots at 1,500 knots
	± 18.0 knots at 180 knots

Table I. (Cont.)

f.	Static Air Temperature $\pm$ 1.25°C over range of -99°C to 60°C
g.	Non-Standard Day Temperature $\pm$ 1.75°C hp <36,090 feet $\pm$ 1.50°C hp >36,090 feet
h.	Maximum Operating Speed $V_{MO} \pm 3.0$ knots
i.	Maximum Operating Speed $V_c = V_{MO}{}^{-0}_{+6}$ knots $M = M_{MO}{}^{-0}_{+0.01}$ Mach
j.	Altitude Hold Engage Error: $\pm$ 2 feet Scale factor conformity: $\pm$ (5 feet + 10 percent of altitude deviation)
k.	Altitude Rate $\pm$ (30 feet per minute or 5 percent of indication whichever is greater).
l.	Mach Hold Engage Error: $\pm$ 0.0005 Mach Scale factor conformity: $\pm$ (-0.0001 Mach + 10 percent Mach deviation)
m.	Airspeed Rate $\pm$ (0.4 knots per second + 10 percent of indicated value)

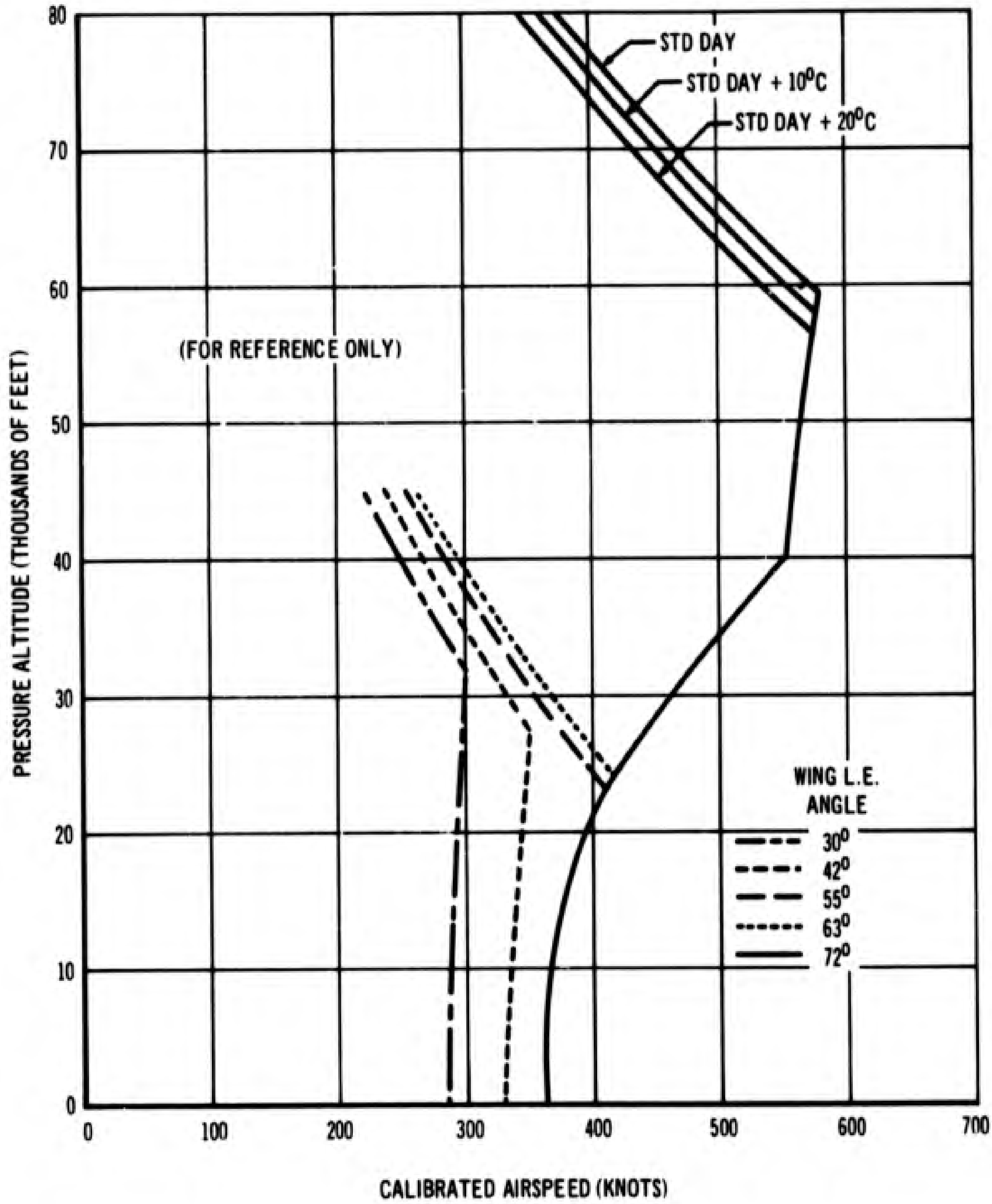


Figure 19. Maximum Operating Speed

Performance only

3.1.1.1.1.9 Slew Rate. The outputs of the computer shall be capable of the following slew rates:

Function	Slew Rate
Altitude	25,000 feet per minute
Calibrated airspeed	600 knots per minute
Mach number	1.0 Mach per minute
Total air temperature	200°C per minute
True airspeed	600 knots per minute
Static air temperature	100°C per minute
Non-standard day temperature	50°C per minute

3.1.1.1.1.10 Threshold. The threshold of the output functions of the computer shall be as follows:

Function	Threshold
Altitude	1 foot at sea level 10 feet at 70,000 feet
Calibrated airspeed	0.2 knots at 100 knots 0.5 knots at 500 knots
Mach number	0.0005 Mach at sea level 0.0015 Mach at 70,000 feet
Total air temperature	0.25°C
Non-standard day temperature	0.25°C
True airspeed	0.5 knots
Static air temperature	0.25°C
Altitude rate	30 feet per minute at sea level
Airspeed rate	0.3 knots per second

3.1.1.1.1.11 Position Error. When the computer is placed in any position, the change in output reading from that obtained with the unit in its normal operating position, shall not exceed 50 percent of the allowable tolerance specified in Table I.

3.1.1.1.2 Total Air Temperature Probe. The total air temperature probe shall provide the capability of measuring the total or stagnation temperature of the airstream. The operating temperature range of the probe shall be minus 70°C to plus 325°C.

3.1.1.1.2.1 Sensing Element. The probe shall contain two electrically independent sensing elements. The sensing elements shall be constructed of pure platinum wire having a nominal resistance value of 500 ohms at a temperature of 0°C.

3.1.1.1.2.2 Deicing. The probe shall be electrically deiced. The probe shall not be damaged by continuous application of heater power under still-air conditions.

3.1.1.1.2.3 Calibration Error. The resistance-temperature calibration of the sensing elements shall conform to the nominal relationship with a tolerance of 0.5°C plus 0.5 percent of the magnitude of the temperature in degrees centigrade.

3.1.1.1.2.4 Recovery Error. At supersonic speeds, the recovery error of the probe shall be between 0.2 and 0.5 percent of the total temperature in degrees Kelvin.

3.1.1.1.2.5 Time Constant. The time constant of the sensing elements shall not exceed 2 seconds at Mach 2.7 and an altitude of 70,000 feet.

3.1.1.1.2.6 Deicing Characteristics. At Mach numbers above 0.7, the error in total temperature measurement because of application of deicing heat shall not exceed degrees centigrade.

3.1.1.1.3 Pitot-Static Installation. The pitot-static installation shall provide the capability of measuring pitot and static pressure and distributing the measured quantities to using equipment. As a minimum requirement, the pitot-static installation shall consist of the following:

Two pneumatically independent primary pitot sources.

Two pneumatically independent primary static sources.

Two secondary pitot sources.

Three pneumatically independent secondary static sources.

3.1.1.1.3.1 Location. The pitot and static sources shall be located to provide pressure measurement with a minimum of interference from the airplane.

3.1.1.1.3.2 Pressure Lag. The total lag (or time) constant of the primary static systems, as measured at the pressure transducers of the air-data computers, shall not exceed seconds at an altitude of 70,000 feet. The total lag constant of the primary pitot systems, as measured at the pressure transducers of the air-data computers, shall not exceed seconds at a pressure of 25 inches of mercury.

3.1.1.1.3.3 Anti-Icing. All probe type sensors shall be electrically heated. The heaters shall be capable of removing accumulated ice and maintaining an ice-free condition sufficient for normal operation, under icing conditions specified in FAR Part 25, Par. 25.1419, and ambient temperatures to 40°C. The probes shall not be damaged by continuous application of heater power under still air conditions. Flush static ports will not be anti-iced.

3.1.1.1.4 Pressure Altitude Indicator. The purpose of the pressure altitude indicator shall be to provide visual indication of pressure altitude based on pneumatic operation and normally corrected with outputs from an air-data computer. The following information shall be displayed on the indicator:

a. Pressure Altitude
b. Selected Barometric Pressure (Inches of Hg and Millibar)
c. Standby Operation Warning

The following controls shall be provided on the face of the indicator:

a. Barometric Pressure Setting Control
b. Normal/Standby Operation Control

3.1.1.1.4.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 20. The indicator shall be designed with the capability of either servo (normal) or pneumatic (standby) operation. In the normal mode, the indicator shall be driven by a servomechanism which positions the pneumatic mechanism to correspond to an electrical signal of corrected pressure altitude supplied by an air-data computer. In the standby mode, the indicator shall operate as a conventional pneumatic altimeter. A warning flag shall be provided which shall become visible on the face of the indicator when the instrument is operated in the standby mode. A switch located on the bezel of the indicator shall enable manual selection of either normal or standby mode of operation.

3.1.1.1.4.2 Leakage. The case leakage shall not exceed 100-feet per minute at a differential pressure of 15 in. of mercury.

3.1.1.1.4.3 Normal Mode Accuracy. With a barometric pressure setting of 29.92 in. of mercury and with appropriate static pressure and electrical signal inputs, the error of the indicator shall not exceed the following limits:

Range (ft)	Error (ft)
-1,000 to 10,000	±10
10,000 to 80,000	±15

The specified tolerance shall include hysteresis, resolution and friction effects.

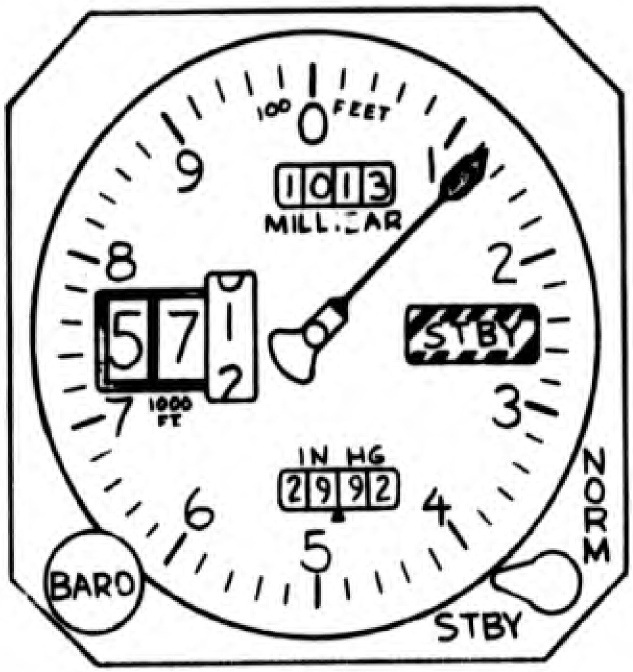


Figure 20. Pressure Altitude Indicator

3.1.1.1.4.4 Standby Mode Performance.

3.1.1.1.4.4.1 Scale Error. With a barometric pressure setting of 29.92 in. of mercury, the scale error of the indicator shall not exceed the following limits:

Altitude (ft)	Scale Error Room Temp.	*Low Temp. (-30°C)
-1,000 to 1,000	±20	±40
2,000	±30	±35
6,000	±40	±40
10,000	±80	±100
20,000	±130	±130
40,000	±230	±230
60,000	±400	±400
80,000	±1,000	±1,000

* Allowable change from room temperature scale error test indication.

3.1.1.1.4.4.2 Hysterisis and After Effect. The difference between the indicator reading taken when the instrument is subjected to pressure simulating increasing altitude and the indicator reading taken when the instrument is subjected to pressures simulating decreasing altitude shall not exceed the following limits:

Altitude (ft)	Tolerance (ft)
25,000	75
20,000	75
Sea Level	30

3.1.1.1.4.4.3 Friction. The difference in the indicator readings taken prior to and after tapping of the instrument case shall not exceed the limits listed below. The integral vibrator shall be disabled during this test.

Altitude (ft)	Tolerance (ft)
1,000	70
5,000	70
10,000	80
20,000	100
30,000	140
40,000	180
50,000	250
60,000	400
80,000	1,200

3.1.1.1.4.4 Failure Warning Requirements. The indicator shall automatically revert to standby mode of operation if any of the following conditions exist:

a. Loss of electrical power to the indicator.

b. Failure of the indicator power supply.

c. Absence of an air-data computer valid signal.

d. A commanded correction (difference between pneumatic indication and air-data computer signal) exceeding   feet.

The flag signal shall be dc voltage and shall be available for external use.

3.1.1.1.5 Radio-Altitude Vertical Speed Indicator. The purpose of the radio-altitude vertical-speed indicator shall be to provide visual indication of the aircraft radio altitude, based on output signals from the radio altimeter, ARINC 552; the aircraft vertical speed, based on output signals from an air-data computer; and a commanded rate of climb, based on output signals from the automatic flight controls system. The following information shall be displayed on the indicator:

Radio Altitude
Minimum-Altitude Warning
Radio-Altitude Failure Warning
Vertical Speed
Vertical-Speed Failure Warning
Rate-of-Climb Command

A control for minimum-altitude warning selection shall be provided on the face of the indicator.

3.1.1.1.5.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 21.

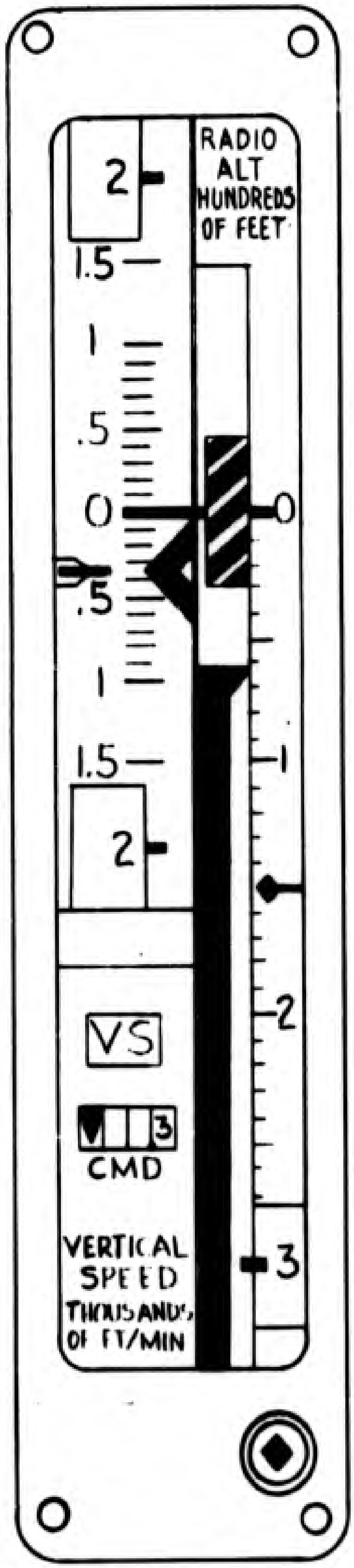


Figure 21. Radio Altitude/Vertical Speed Indicator

3.1.1.1.5.1.1 Vertical Speed. Vertical speed shall be presented by a moving index which is read against a fixed scale between the range of minus 1,500 to plus 1,500 fpm. At vertical speeds exceeding plus or minus 2,000 fpm, the moving index shall stop and readout of vertical speed shall be presented by a numerical display which is read against the fixed index. The full-scale range of the vertical speed display shall be plus or minus 20,000 fpm. A control, accessible from the face of the indicator, shall be provided to enable zero adjustment of the vertical speed display.

3.1.1.1.5.1.2 Radio Altitude. Radio altitude shall be presented by a moving tape which is read against a fixed scale over the range of -20 to +300 feet. At altitudes above 300 feet, the tape shall be stationary and display is provided by a numerical readout. The full-scale range of the radio altitude display shall be -20 to +2,500 feet.

3.1.1.1.5.1.3 Minimum Altitude. A minimum altitude cursor and set knob shall be provided. The cursor shall be manually set at any altitude between 0 to 270 feet. A switch shall be included in the indicator which shall be actuated whenever the indicated radio altitude is at or below the minimum set value.

3.1.1.1.5.1.4 Rate-of-Climb Command. Rate-of-Climb Command shall be presented by a moving index which is read against a fixed scale between the range of -1,000 to +1,500 fpm. At commands exceeding ±2,000 fpm, the moving index shall disappear and readout of rate-of-climb command shall be presented by a numerical counter. The full-scale range of the rate-of-climb pointer and counter shall be ±20,000 fpm. The control of the rate-of-climb command shall be from a remote source. The indicator shall generate an error signal proportional to the difference between the actual vertical speed and the commanded rate-of-climb. The error signal shall be utilized, either within the autopilot to control the aircraft or within the remote selector to synchronize its commanded output to the vertical speed, depending on the selected mode of the autopilot.

3.1.1.1.5.2 Accuracy.

3.1.1.1.5.2.1 Vertical Speed. The vertical speed indication shall correspond to the electrical input with maximum errors as listed below:

Rate (fpm)	Error (fpm)
0	±25
±1,000	±50
±2,000	±100
±4,000	±100
±8,000	±200
±20,000	±500

3.1.1.1.5.2.2 Radio Altitude. The radio altitude indication shall correspond to the electrical input with maximum errors as listed below:

Altitude (ft)	Error (fpm)
-20	± 2.0
+200	± 2.0
+500	± 5.0
+2,500	± 50.0

3.1.1.1.5.2.3 Minimum Altitude. The minimum-altitude warning switch shall be actuated when the indicated radio altitude is within ±5.0 feet of the preset minimum altitude.

3.1.1.1.5.3 Threshold. The minimum change of electrical input required to produce a detectable change of instrument indication shall not exceed the following levels:

Function	Threshold
Vertical Speed	10 fpm
Radio Altitude	1.0 foot

3.1.1.1.5.4 Radio Altitude Slew Rate. The radio altitude indicator servo shall be capable of a slew rate of ± 6,000 fpm.

3.1.1.1.5.5 Radio-Altitude Servo Damping. The radio-altitude servo shall be damped so the overshoot in response to a step input shall not exceed 10 percent.

3.1.1.1.5.6 Vertical-Speed Servo Performance. The vertical-speed servo shall be designed to respond in an exponential manner to a step input of 2,000 fpm. The time constant of the servo under these conditions shall be adjustable over a range of one to six seconds. Adjustments for setting the desired response shall be provided in the vertical-speed servo amplifier module.

3.1.1.1.5.7 Failure Warning Requirements.

3.1.1.1.5.7.1 Vertical Speed. A failure warning flag shall be provided which shall become visible on the vertical-speed scale if any of the following conditions exist:

a. Loss of electrical power to the indicator
b. Failure of the indicator power supply
c. Excessive vertical-speed servo null voltage

The flag signal shall be dc voltage and shall be available for external use.

3.1.1.1.5.7.2 Radio Altitude. A failure warning flag shall be provided which becomes visible on the altitude scale if any of the following conditions exist:

a. Loss of electrical power to the indicator
b. Failure of the indicator power supply
c. Excessive radio-altitude servo null voltage
d. Absence of a radio altimeter valid signal

The flag signal shall be dc voltage and shall be available for external use.

3.1.1.1.5.8 Monitoring. Electrical outputs of vertical speed and radio altitude shall be provided for purposes of comparison monitoring.

3.1.1.1.6 Airspeed-Mach-Air Temperature Indicator. The purpose of the airspeed-mach-air temperature indicator shall be to provide visual indication of airspeed based on pneumatic operation normally corrected with outputs from an air-data computer and mach and air temperature (static, total, and non-standard day) based on outputs from an air-data computer. The following information shall be displayed on the indicator:

Calibrated/Indicated Airspeed
Standby Operation Warning
Calibrated Airspeed Command
Maximum Operating Speed
Mach Number
Mach Number Failure Warning
Air Temperature (static, total, and non-standard day)
Air Temperature Failure Warning (static, total, and non-standard day)

The following controls shall be provided on the face of the indicator:

Normal/Standby Operation Control
Air-Temperature Selector Switch

3.1.1.1.6.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 22.

3.1.1.1.6.1.1 Airspeed. The airspeed indicating mechanism shall be designed with the capability of both servo (normal) or pneumatic (standby) operation.

3.1.1.1.6.1.1.1. Airspeed Pointer. When operating in the normal mode, the airspeed pointer shall be driven by a servomechanism which positions the pneumatic mechanism to correspond to an electrical signal of calibrated airspeed supplied by an air-data computer. In the standby mode, the airspeed pointer shall be positioned directly by the pneumatic mechanism.

3.1.1.1.6.1.1.2 Airspeed Counter. During normal mode operation, the airspeed counter shall be positioned to display calibrated airspeed as supplied by an air data computer. In the standby mode, the airspeed counter shall be masked by a flag.

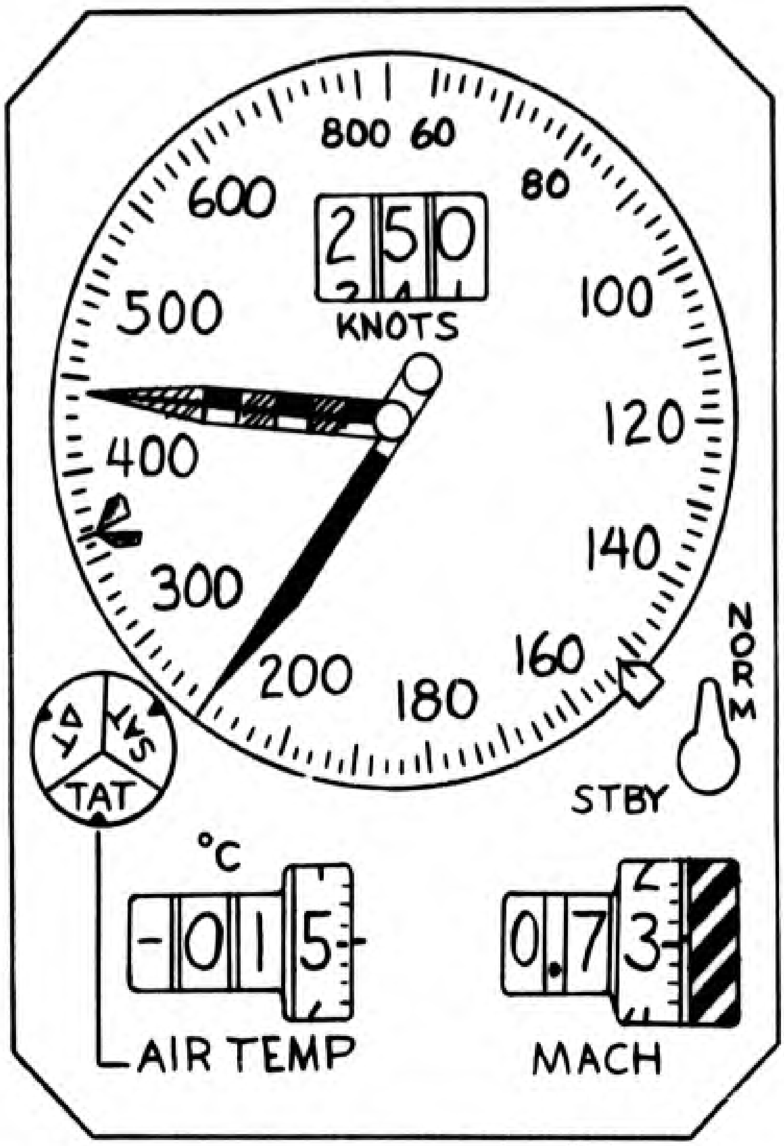


Figure 22. Airspeed-Mach-Air Temperature Indicator

3.1.1.1.6.1.1.3 Airspeed Command. During normal operation, the airspeed command cursor shall display airspeed command which has been manually selected at a remote source.

3.1.1.1.6.1.1.4 Maximum Operating Airspeed (V_{MO}) Pointer. The V_{MO} pointer shall display maximum operating airspeed as supplied by an air-data computer.

3.1.1.1.6.1.1.5 Mode Selector Switch. The mode selector switch shall enable manual selection of the mode of operation of the airspeed indicating mechanisms.

3.1.1.1.6.1.1.6 Airspeed Reference Cursor. The airspeed reference cursor shall be designed to be easily affixed to and positioned on the exterior of the indicator bezel. The operator may elect to affix more than one cursor for various reference speeds.

3.1.1.1.6.1.2 Mach Number. The Mach number indication mechanism shall accept an electrical input of Mach number supplied by an air data computer. To effectively present Mach rate to the pilot, there shall be a drum with barber pole markings adjacent to the continuous moving hundredth digit and geared to it with a 10:1 ratio.

3.1.1.1.6.1.3 Air Temperature. The air temperature mechanism shall accept an electrical input of air temperature supplied by an air-data computer. A three-position switch shall enable selection of either static, total, or non-standard day temperature.

3.1.1.1.6.2 Accuracy.

3.1.1.1.6.2.1 Airspeed Pointer (Normal Mode). With appropriate pitot and static pressure and electrical signal inputs applied, the error of the indication of the airspeed pointers shall not exceed the following values:

Range, knots	Tolerance, knots
60 to 200	±0.5
200 to 600	±1.5
600 to 800	±3.0

3.1.1.1.6.2.2 Airspeed Counter (Normal Mode). The airspeed counter indication shall correspond to the electrical input signal with a maximum error of ± 0.5 knots. The counter readings shall agree with the airspeed pointer readings within the following values:

Range, knots	Tolerance, knots
60 to 200	1.0
200 to 400	2.0
600 to 800	3.5

3.1.1.1.6.2.3 Command Airspeed (Normal Mode). The indication of the command airspeed cursor shall correspond to the electrical input with a maximum error of ± 1 knot.

3.1.1.1.6.2.4 Maximum Operating Airspeed (Normal Mode). The indication of the maximum operating airspeed pointer shall correspond to the electrical input with a maximum error of ± 1 knot.

3.1.1.1.6.2.5 Airspeed Indication Scale Error (Standby Mode). The scale error of the airspeed indication shall not exceed the following limits:

Airspeed (knots)	Scale Error (knots)
50	±5.0
100	±2.0
150	±2.5
200	±3.0
400	±5.0
600	±7.0
800	±9.0

3.1.1.1.6.2.6 Airspeed Indication Friction (Standby Mode). The difference between the indicator readings taken prior to and after tapping of the instrument case shall not exceed two knots.

3.1.1.1.6.2.7 Mach Number. The Mach number indication shall correspond to the electrical input signal with a maximum error of plus or minus 0.005 Mach.

3.1.1.1.6.2.8 Air Temperature. The air temperature indication shall correspond to the electrical input signal with a maximum error of ± 0.25°C.

3.1.1.1.6.3 Follow-Up Rate. The indicator servos shall be capable of the following follow-up rates:

Function	Rate
Airspeed (Normal mode-pressure inputs and electrical signal inputs changing in unison)	300 knots/min
Mach Number	1.0 Mach/min
Air Temperature	300°C/min
Command Airspeed	600 knots/min
Maximum Operating Airspeed	300 knots/min

3.1.1.1.6.4 Damping. All indicator servos shall be damped so the overshoot in response to a step input shall not exceed 10 percent.

3.1.1.1.6.5 Failure Warning Requirements.

3.1.1.1.6.5.1 Calibrated Airspeed. The indicator shall automatically revert to standby mode of operation if any of the following conditions exist:

a. Loss of electrical power to the indicator

b. Failure of the indicator power supply

c. Absence of an air data computer valid signal

d. A command correction (difference between pneumatic indication and air data computer signal) exceeding knots

e. The flag signal shall be a dc voltage and shall be available for external use

3.1.1.1.6.5.2 Mach Number. A failure warning flag shall appear if any of the following conditions exist:

a. Loss of electrical power to the indicator

b. Failure of the Mach-servo power supply

c. Absence of an air-data computer valid signal

d. Excessive Mach-servo null voltage

e. The flag signal shall be a dc voltage and shall be available for external use.

3.1.1.1.6.5.3 Air Temperature. A failure warning flag shall appear if any of the following conditions exist:

a. Loss of electrical power to the indicator

b. Failure of the air-temperature servo power supply

c. Absence of an air-data computer valid signal

d. Excessive air-temperature servo null voltage

e. The flag signal shall be a dc voltage and shall be available for external use.

3.1.1.1.6.6 Monitoring.

3.1.1.1.6.6.1 Calibrated Airspeed. An electrical output signal for comparison monitoring shall be provided.

3.1.1.1.7 Altitude Reporting Indicator. The purpose of the altitude reporting indicator shall be to provide visual indication of altitude based on the altitude reporting digital output signals of an air-data computer.

3.1.1.1.7.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 23. Readout of altitude shall be provided in 100-foot increments over the range of -1,000 to +80,000 ft. The use of segmented incandescent lamps, segmented electroluminescent, magnetic wheels or equivalent may be used in the design of this indicator.

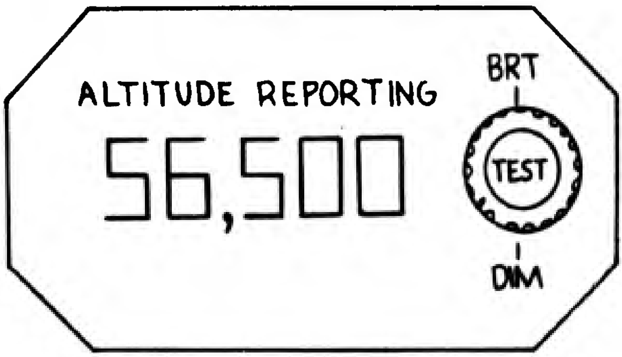


Figure 23. Altitude Reporting Indicator

A control for varying the intensity of the indicators, if required, shall be provided. Located concentrically with the intensity control shall be a push-button test switch. The test switch shall be employed to check that all segments of the thousands and hundreds numerics are operating properly.

3.1.1.1.7.2 Slew Rate. The indicator shall be capable of a slew rate of 25,000 fpm.

3.1.1.1.7.3 Failure Warning Requirements. When a failure of the altitude output of the air-data computer is detected, the encoder output will be disabled through opening of the common return lead of the encoder. When this occurs, the indicator shall display all "zeros".

3.1.1.1.8 True-Airspeed (TAS) Indicator. The purpose of the TAS indicator shall be to provide visual indication of the true airspeed based on output signals from an air-data computer. The following information shall be displayed on the indicator.

True Airspeed

True Airspeed Failure Warning

A control for selection of either of two input sources shall be provided on the face of the indicator.

3.1.1.1.8.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 24.

3.1.1.1.8.1.1 True Airspeed. True airspeed (TAS) shall be presented on a four-digit counter. A warning flag shall mask the counter when a failure of the computing or indicating mechanism is detected.

3.1.1.1.8.1.2 Input Selector Switch. A switch shall be provided to enable selection of either of two air-data computers as the source of input data, including power input, to the indicator.

3.1.1.1.8.2 Accuracy. The TAS indication shall correspond to the electrical input with a maximum error of ± 1 knot.

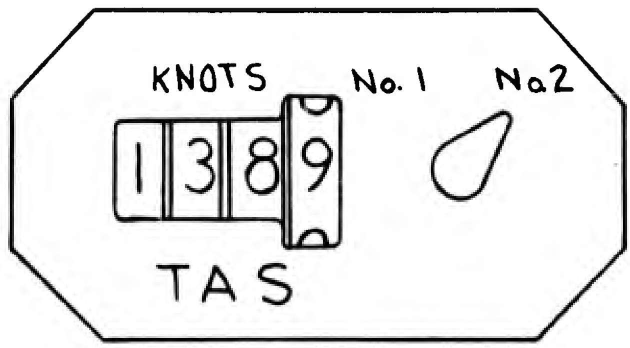


Figure 24. True Airspeed Indicator

3.1.1.1.8.4 Damping. The TAS servo shall be damped so the overshoot in response to a stop input shall not exceed 10 percent.

3.1.1.1.8.5 Failure Warning Requirements. The TAS failure warning flag shall appear in the counter window if any of the following conditions exist:

a. Loss of electrical power to the indicator
b. Failure of the TAS servo power supply
c. Absence of an air-data computer valid signal
d. Excessive TAS servo null voltage

3.1.1.1.9 Attitude Director Indicator. The primary functions of the attitude director indicator (ADI) shall be to provide visual indications of aircraft attitude, director commands, and landing situation information, based on the output signals of remote sensors and remote computing equipment. The following information shall be displayed on the indicator.

a. Pitch attitude
b. Bank angle
c. Attitude information failure warning
d. Pitch command
e. Roll command
f. Pitch-and-roll command-data failure warning
g. Speed command
h. Speed command data failure warning
i. Localizer deviation
j. Localizer failure warning
k. Vertical flight path deviation
l. Vertical flight path deviation warning
m. Radio altitude
n. Radio altitude data failure warning
o. Turn rate
p. Turn rate information failure warning
q. Lateral acceleration
r. De-crab indication
s. Master warning for comparator warning and approach gate monitor
t. Indications of spit axis autopilot operation (pitch or roll axis disengaged).

The following controls shall be provided on the face of the indicator:

a. Pitch attitude trim control (provisions only)
b. Flight director command bars ON-OFF switch

3.1.1.1.9.1 Method of Presentation and Mechanization. Typical display arrangements are shown in Figs. 25, 26 and 27. The method used for mechanizing the pitch, roll and command displays varies with each configuration shown. The selection of a final display configuration will be based upon evaluation of vendor proposals.

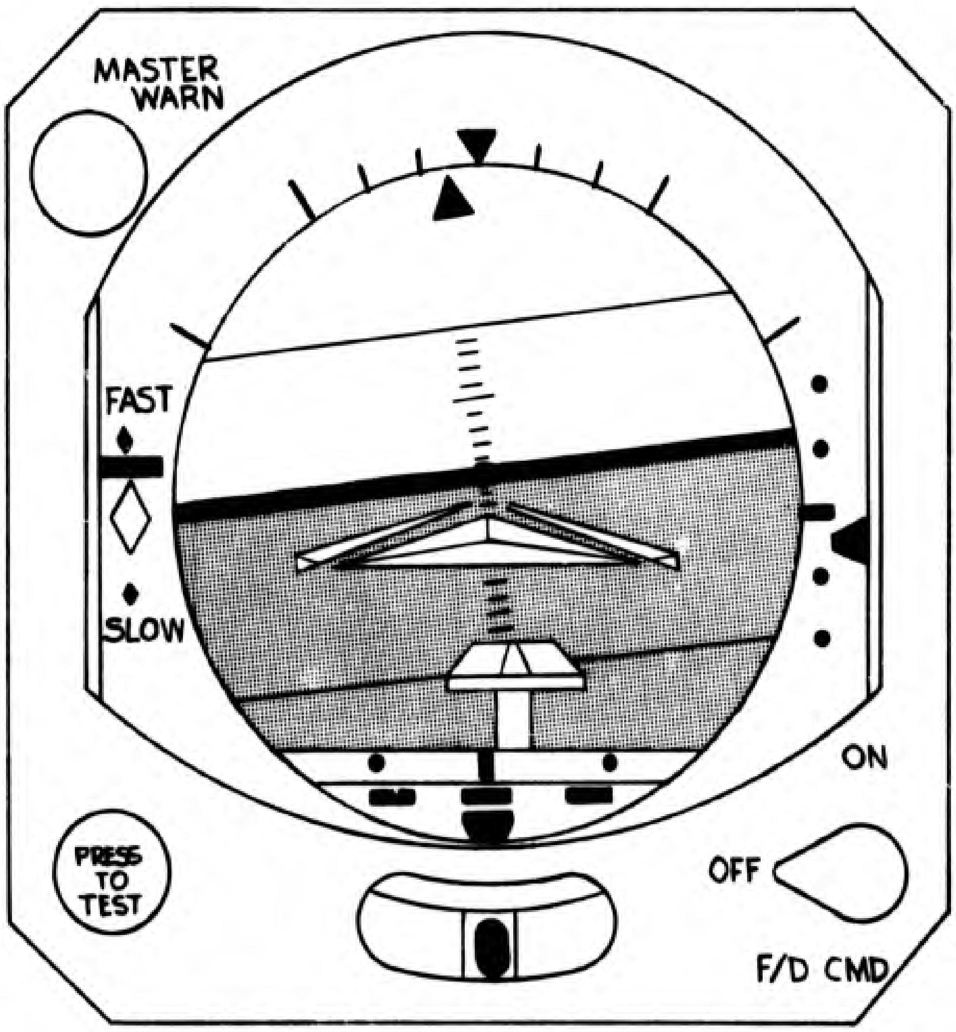


Figure 25. Attitude Director Indicator (Roller Blind Attitude-Integrated Command Configuration)

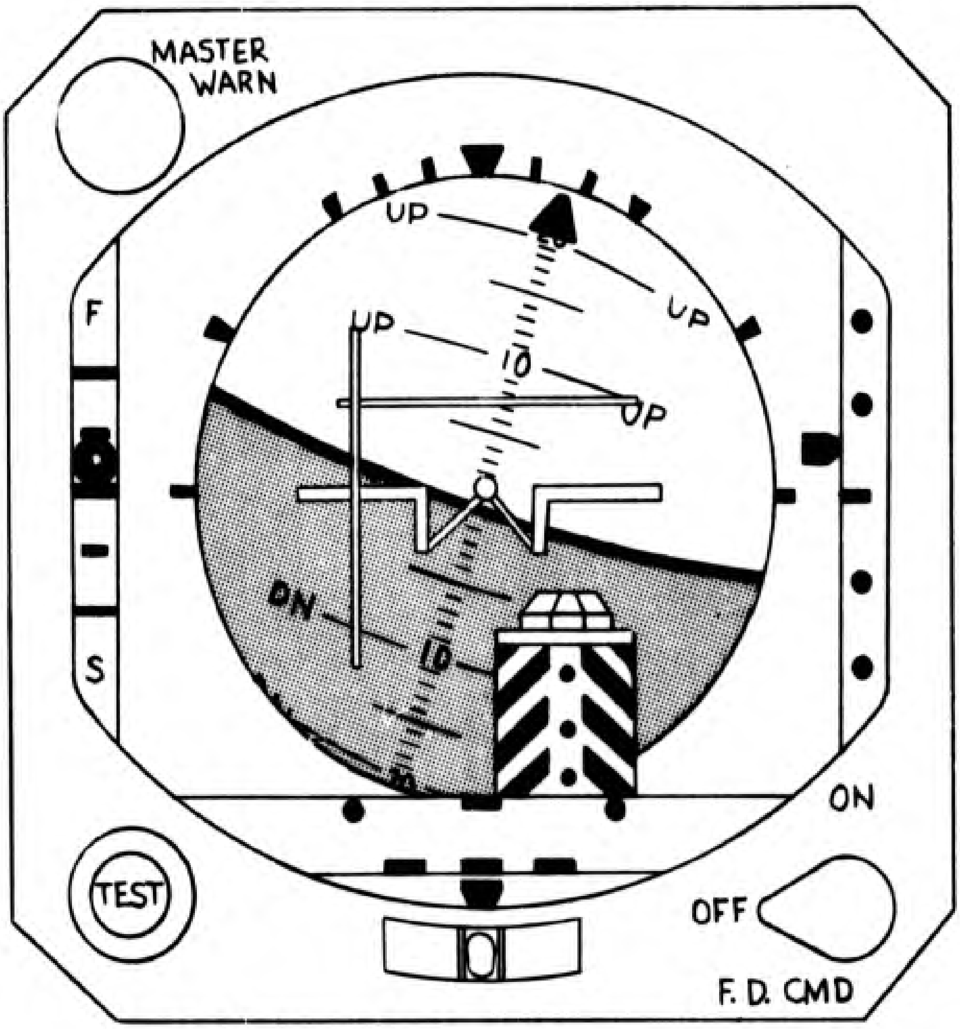
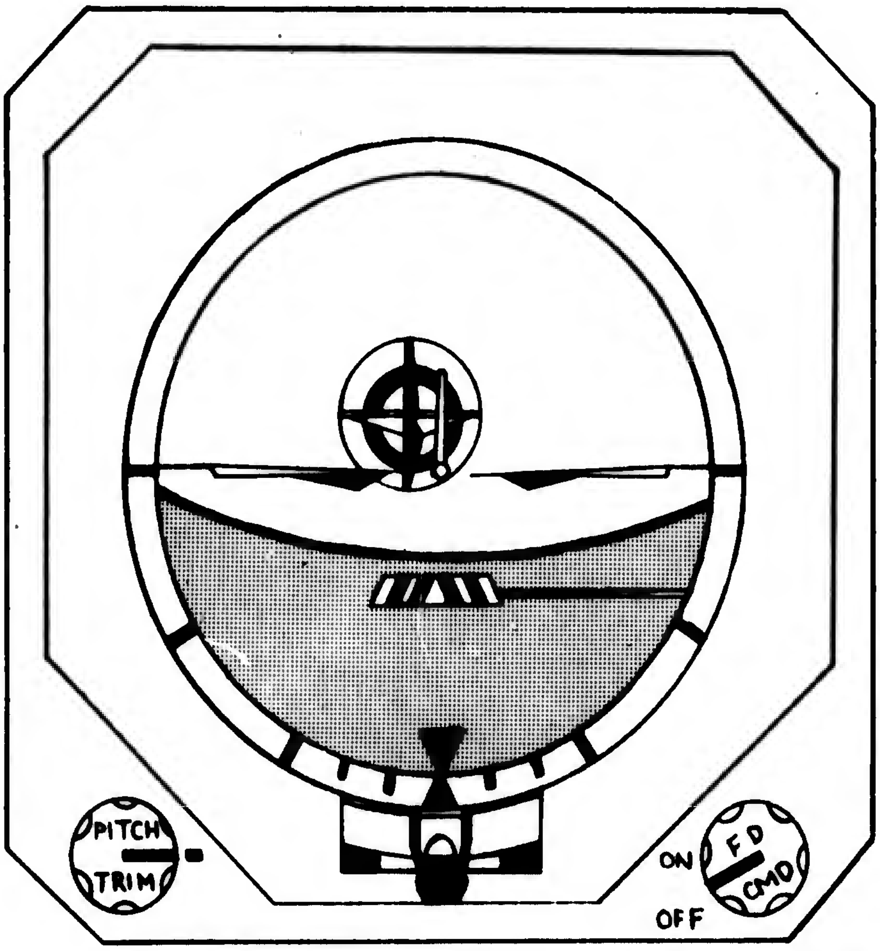


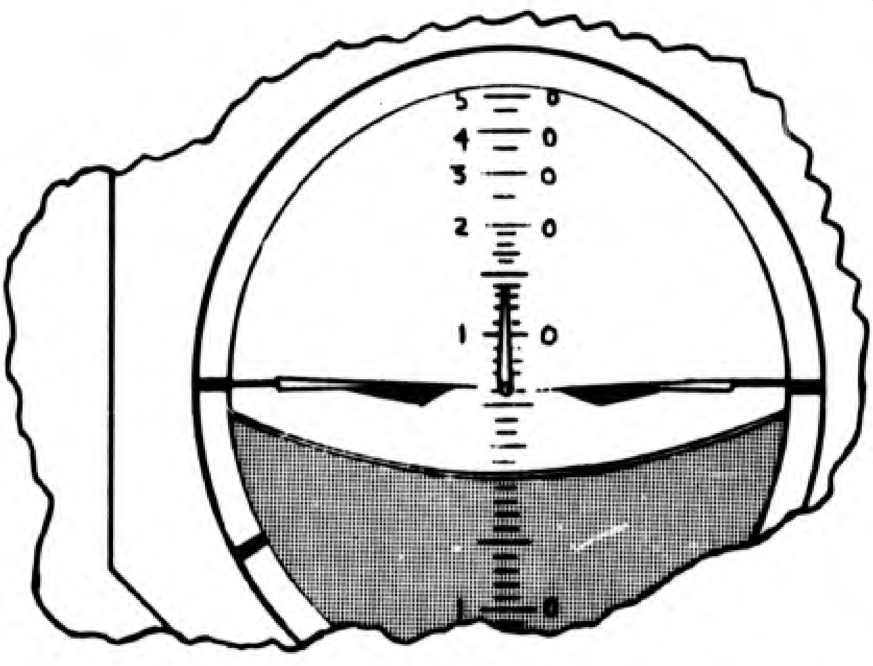
Figure 26. Attitude Director Indicator (Spherical Attitude-Separate Roll and Pitch Command Configuration)

3.1.1.1.9.1.1 Attitude Display. The attitude display shall be distinctively colored, light-blue representing sky, black representing earth, and a bold white line separating the two sections representing the horizon. Pitch attitude shall be clearly subdivided in increments of 1° through ± 20°. In addition, pitch attitude shall be clearly marked in increments of five degrees and numbered in ten degree increments from 0 to ± 40 degrees. Positive and negative pitch angles from 40° to 90° shall be clearly and unambiguously discernable. Throughout the length of the usable pitch scale, the ten degree divisions shall be of sufficient length to serve as a displaced horizon-like reference when the true-horizon line is out of view at extreme pitch attitudes. The major bank angle graduations shall be at 0°, 30°, 60°, 90° and 180°. Minor bank angle graduations shall be at 10° and 20°.

3.1.1.1.9.1.2 Pitch Sensitivity. Pitch sensitivity from -20° to +20° shall shall be 0.075 inch per degree. Attitude presentation may be of the spherical or roller-blind type. If the spherical type is proposed, scale compression from 20° to 90° shall be allowed.



APPROACH MODE



CRUISE MODE

Figure 27. Attitude Director Indicator (Spherical Attitude-Integrated Command Configuration)

3.1.1.1.9.1.3 Command Display. Display elements shall be provided for computed roll-and-pitch commands. The command display elements shall be of a type that does not compromise the readability of pitch attitude when the commands are zeroed or nearly zeroed. A display element shall be provided for speed error or speed command for use primarily during landing approach. A display element shall be provided for de-crab command. The speed command and the de-crab command symbols shall be readable by the pilot within his foveal vision without eye shift from the roll and pitch command display. The roll, pitch and speed command shall utilize as their signal source the autopilot/flight director computer. The de-crab electrical input shall be a signal proportional to the difference between the selected runway heading and the aircraft heading. Means shall be provided to remove the command displays and associated warning flags from view when not in use.

3.1.1.1.9.1.4 Gate Display. A display of localizer deviation and vertical path deviation shall be provided. The deviation information shall be integrated with the other display elements in such a manner that the pilot may easily perceive the relationship of the aircraft to the ILS gate or corridor from which a safe landing can be accomplished. The gate display shall be readable by the pilot within his foveal vision without eye shift from the command display. Means shall be provided to remove the gate display from view when not in use.

3.1.1.1.9.1.5 Radio Altitude Display. Radio altitude shall be symbolically displayed. The full-scale range of the altitude indicator shall be 200 feet, and the motion shall be linear with altitude. Over the indicating range, the altitude symbol shall be discernable by the pilot with a minimum of eye shift from the command and displacement displays. The design of the altitude symbol shall be such that the pilot is apprised of localizer deviation while scanning the altitude display.

3.1.1.1.9.1.6 Annunciators and Master Warning. An amber master warning light shall be incorporated at the upper left corner of the instrument face. The design of the master warning cancellation provisions shall be coordinated with the design of the comparator warning and approach gate monitor system. Provisions shall be made in the design for indicator to apprise the pilot of the operation of the autopilot in a split-axis mode, and as a reminder of which axis is to be controlled manually.

3.1.1.1.9.1.7 Colors. The mask and lower portion of the sphere or roller blind shall be dull black. Graduations and numerals shall be matte white with the exception of those on the upper portion of the sphere or roller blind. Consideration shall be given to methods of providing adequate contrast between the light blue of the upper portion of the sphere or roller blind and the associated pitch graduations and numerals. Warning flags carrying a legend shall be red Nazder lacquer DL 101 or equivalent with dull black letters.

3.1.1.1.9.2 Accuracy. The indicator shall correspond to the electrical input with maximum errors listed below:

Function	Range	Maximum Static Error
Pitch Attitude	0 ± 20°	0.25°
	± (20 to 90°)	0.50°
	0°	0.25°
	± (10 to 20°)	0.50°
	± (30 to 180°)	0.75°
Pitch Command	To be determined	
Roll Command	To be determined	
Speed Command	0 speed error	30 microamperes
	Full scale	80 microamperes
Localizer Deviation	On Course	4 microamperes
	1° beam	6 microamperes
Vertical Flight Path Deviation	On Course	4 microamperes
	Half-scale	6 microamperes
	Full-scale	12 microamperes
Radio Altitude	0 to 200 ft	10%
Turn Rate	0°/sec.	0.10°/sec.
	1.5°/sec.	0.12°/sec.
	3.0°/sec.	0.25°/sec.

3.1.1.1.9.3 Slew Rate. The indicator servos shall be capable of the following slew rates:

Function	Rate
Pitch Attitude	40°/sec.
Bank Attitude	60°/sec.
Pitch-and-Roll Command	To be determined

3.1.1.1.9.4 Threshold. The minimum change of electrical input required to produce a detectable change of instrument indicator shall not exceed the following:

Function	Threshold, °
Pitch Attitude	0.10
Bank Attitude	0.10
Pitch-and-Roll Command	0.10

3.1.1.1.9.5 Servo Damping. The instrument servos shall be damped so the overshoot in response to a stop input shall not exceed 10 percent.

3.1.1.1.9.6 Self Test. A manually actuated self-test mode shall be provided to enable a preflight checkout of the attitude portion of the indicator and bulb condition of the annunciator lights. The attitude failure flag shall be in view during the self-test mode.

3.1.1.1.9.7 Failure Warning Requirements.

3.1.1.1.9.7.1 Servo-Operated Functions. A failure warning flag shall be provided for each servo-operated function which shall become visible if any of the following conditions exist in the respective servo systems:

a. Loss of electrical power to the indicator
b. Failure of servo power supply
c. Absence of an input valid signal
d. Excessive servo null voltage

The flag signals shall each be a dc voltage and shall be available for external use.

3.1.1.1.9.7.2 Meter Movement Operated Functions. A failure warning flag shall be provided for each meter movement operated function which shall become visible in the absence of the respective input valid signals.

3.1.1.1.9.8 Monitoring. Electrical outputs of pitch-and-bank attitude and pitch-and-roll command shall be provided for purposes of comparison monitoring.

3.1.1.1.10 Horizontal Situation Indicator. The primary functions of the horizontal situation indicator (HSI) shall be to display aircraft heading and to integrate the related displays of course deviation or cross track deviation, selected course or desired track angle, and pictorial representation of the navigation situation. The following information shall be displayed on the indicator:

a. Radio/Navigation Selection Annunicator
b. Heading (true or magnetic)
c. Annunciation of type of heading information being displayed (true or magnetic)

d. Heading failure warning (may be combined with above annunicator)
e. Selected heading
f. Selected course (or desired track angle during inertial navigation)
g. Course deviation (VOR, localizer, or inertial cross-track)
h. Navigation failure warning
i. VOR ambiguity (to-from)
j. Glide slope deviation
k. Glide slope failure warning
l. DME 1
m. DME 2
n. Radio Bearing 1
o. Radio Bearing 2

3.1.1.1.10.1 Method of Presentation and Mechanization. The display arrangement shall be essentially as shown in Fig. 28.

3.1.1.1.10.1.1 Heading. Repeated heading shall be displayed by a servo-driven dial in conjunction with a pointer-shaped index. The dial shall have five degree graduation marks. Additional fixed index markers shall be located at 45° and 90° on each side of the heading index and a bar-shaped reciprocal heading mark shall extend over the bottom edge of the dial. The heading function and its respective servo and amplifier shall be electrically isolated from the other instrument functions.

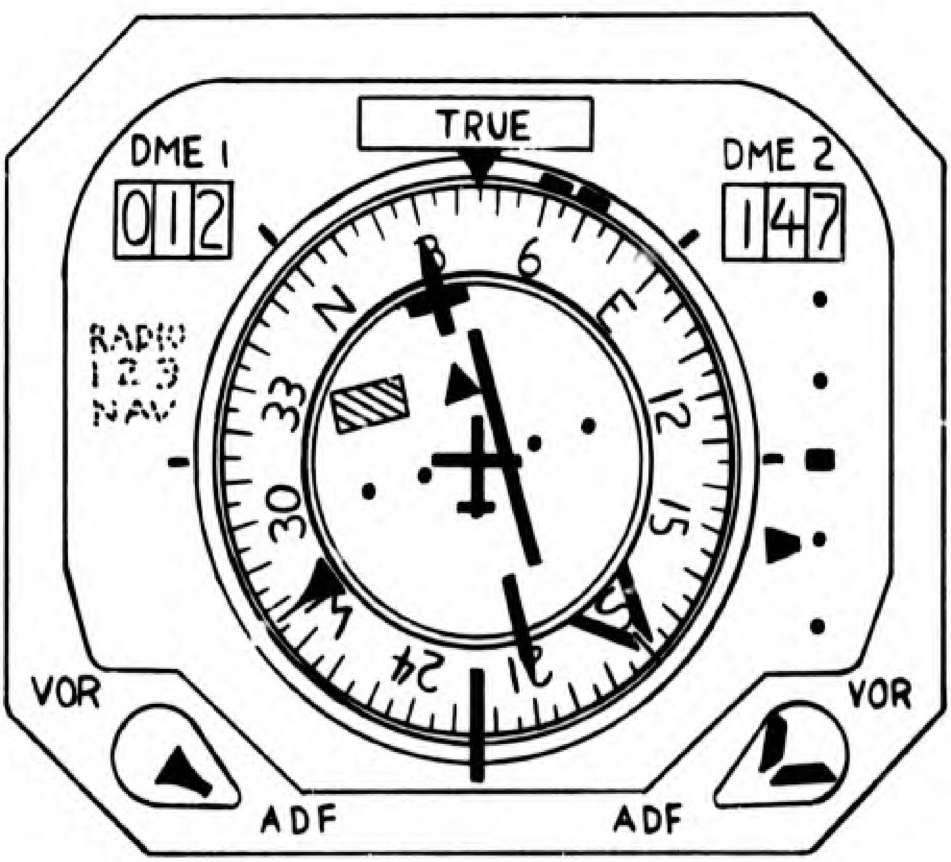


Figure 28. Horizontal Situation Indicator

3.1.1.1.10.1.2 Heading Select Cursor.

a. Radio Navigation Mode. The cursor shall display selected heading and shall be controlled from a remote source. The electrical input shall be a signal proportional to the difference between selected heading and aircraft heading.

b. Inertial Navigation Mode. The cursor shall display present tract angle. The electrical input shall be a signal proportional to aircraft drift angle.

3.1.1.1.10.1.3 Course Select Cursor.

a. Radio Navigation Mode. The cursor shall display selected course and shall be controlled from a remote source. The electrical input shall be a signal proportional to the difference between selected course and aircraft heading. The VOR course resolver shall be located at the remote course select control.

b. Inertial Navigation Mode. The cursor shall display desired track angle (desired course). Two input signal configurations are possible depending on the type of outputs selected for the inertial navigation system by the ARINC Subcommittee. The alternate configurations are:

(1) The electrical input shall be a signal proportional to the sum of track-angle error and drift angle. (Preferred), or

(2) The electrical input shall be a signal proportional to track-angle error. The drift-angle signal used to position the selected heading cursor and the track-angle error signal shall be employed to derive track-angle error plus drift-angle error plus drift angle for controlling the course cursor.

3.1.1.1.10.1.4 Radio Bearing. A servo-driven arrow shall indicate against the heading dial the bearing to the radio station selected on the No. 1 radio receiver. A servo-driven open triangle shall indicate against the heading dial the bearing to the radio station selected on the No. 2 radio receiver. Means shall be provided to automatically orient the bearing pointers to a non-obstructive position when they are not in use.

3.1.1.1.10.1.5 Course Deviation. Deviation from a desired course (VOR/LOC or inertial) shall be indicated by a meter-driven bar moving across a conventional four-dot scale affixed to the rotating mask. A fixed miniature airplane symbol shall provide a pictorial representation of the aircraft relationship to the desired course line.

3.1.1.1.10.1.6 Glide Slope Deviation. Glide slope deviation shall be displayed by a triangular-shaped pointer moving against a fixed scale at the right-hand edge of the instrument face. The glide slope shall have two dots above and two dots below the on-glide-slope index. Means shall be provided to remove the glide slope display, including the warning flag, from view when not in use.

3.1.1.1.10.1.7 DME. Two separate, three-digit DME counters having numerals 0.25 in high shall be included. Separate synchros shall be used to drive units, tens, and hundreds readouts. The counter in the upper left corner of the instrument face shall be designated "DME 1", and the counter in the upper right corner of the instrument face shall be designated "DME 2".

3.1.1.1.10.1.8 Annunciators and Warnings.

3.1.1.1.10.1.8.1 A course deviation failure warning flag shall be provided.

3.1.1.1.10.1.8.2 A VOR To-From display shall be provided. A triangular-shaped pointer shall appear to the rotating mask in front of the miniature airplane symbol to indicate the aircraft flying to the VOR station; a similar pointer shall appear behind the miniature airplane symbol to indicate the aircraft flying from the VOR station. "Over the station" and radio OFF or failed information shall be indicated by the disappearance of the pointers behind the mask.

3.1.1.1.10.1.8.3 Heading failure warning shall be provided by a flag near the top center of the instrument face.

3.1.1.1.10.1.8.4 An annunciator near the top center of the instrument face shall indicate the type of heading information being displayed, TRUE or MAGNETIC. This annunciator may be combined with the heading failure warning if desired.

3.1.1.1.10.1.8.5 Glide slope failure shall be indicated by the appearance of a flag. The flag shall obscure at least the center portion of the glide slope scale.

3.1.1.1.10.1.8.6 An annunciator to the left of the instrument face shall indicate type of deviation and course information being displayed; RADIO 1 or 2, NAV 1 or 2 or 3.

3.1.1.1.10.1.8.7 A shutter shall partially obscure each DME counter when the appropriate bias voltage is lacking.

3.1.1.1.10.1.9 Colors. Dials and masks shall be dull black. Scales, numerals and legends shall be matte white. If advantageous for contrast, colors may be used for pointers and indices. Warning flags carrying a legend shall be red Nazder lacquer DL 101 or equivalent with dull black letters. Warning flags not carrying a legend shall be cross-hatched red and white.

3.1.1.1.10.2 Accuracy. The indicator shall correspond to the electrical input with maximum errors as listed below:

Function	Range	Maximum Static Error
Heading		0.5°
Course Deviation	On Course	4 microamperes
	Half-Scale	6 microamperes
	Full-Scale	12 microamperes
DME (combined indicator of units, tens and hundreds counter)		0.1 nmi
Radio Bearing		1.0°

3.1.1.1.10.3 Slew Rate. The indicator servos shall be capable of the following slew rates:

Function	Rate, °/sec.
Heading	30
Selected Heading	30
Selected Course	30
Radio Bearing	60

3.1.1.1.10.4 Threshold. The maximum change of electrical input required to produce a detectable change of instrument indication shall not exceed the following:

Function	Threshold
Heading	0.10°
Selected Heading	0.20°
Selected Course	0.20°
DME	0.05 nmi
Radio Bearing	0.20°

3.1.1.1.10.5 Damping.

3.1.1.1.10.5.1 Servo Damping. The instrument servos shall be damped such that the overshoot in response to a stop input shall not exceed 10 percent.

3.1.1.1.10.5.2 Deviation Indicator Damping. The course and glide slope deviation mechanisms shall be damped such that no overshoot occurs in response to a stop input.

3.1.1.1.10.6 Self Test. Annunciator lights shall have a test feature to indicate bulb condition. If advantageous, a bearing selector knob may be integrated with the annunciator light test feature.

3.1.1.1.10.7 Failure Warning Requirements.

3.1.1.1.10.7.1 Heading Failure Warning. A failure warning flag shall be provided for heading which shall become visible if any of the following conditions exist:

a. Loss of electrical power to the indicator
b. Failure of servo power supply
c. Absence of an input valid signal
d. Excessive servo null voltage

The flag signal shall be a dc voltage and shall be available for external use.

3.1.1.1.10.7.2 Navigation, Glide Slope and DME Failure Warning. Navigation, glide slope and DME failure warning flags shall be provided which shall become visible in the absence of the respective input valid signal.

3.1.1.1.10.8 Monitoring. An electrical output of heading shall be provided for purposes of comparison monitoring.

3.1.1.1.11 Standby Attitude Indicator. The functions of the standby attitude indicator shall be to display aircraft pitch attitude and bank angles.

3.1.1.1.11.1 Method of Presentation and Mechanization. A typical display arrangement is shown in Fig. 29. Pitch and bank information shall be presented by a servo-positioned attitude sphere of roller blind. The attitude sphere shall have full 360° freedom in roll and a range of 0 ± 90° in pitch. The attitude display shall be distinctively colored, light-blue representing sky, black representing earth, and a bold white line separating the two sections representing the horizon. Pitch attitude shall be clearly marked in increments of 5° up to 20° and in increments of 10° from 20° to 90°. A failure warning flag for the attitude display shall be provided. The attitude display shall be consistent in form, color, and design with the primary display.

3.1.1.1.11.2 Accuracy. The indicator shall correspond to the electrical input with maximum errors as listed below:

Function	Range	Maximum Static Error, °
Pitch	0 ± 20°	± 0.50
	± (20 to 90°)	± 1.0
Bank	0 ± 30°	± 0.50
	± (30 to 180°)	± 1.0

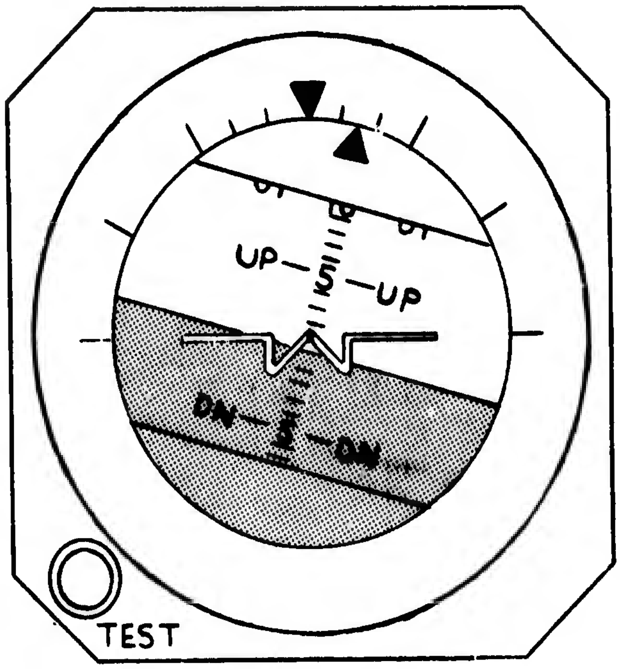


Figure 29. Standby Attitude Indicator

3.1.1.1.11.3 Slew Rate. The slew rate capability of the indicator servos shall meet or exceed the following levels:

Function	Slew Rate, °/sec.
Pitch	40
Bank	60

3.1.1.1.11.4 Threshold. The minimum change of electrical input required to produce a detectable change of instrument indication shall not exceed the following:

Function	Threshold, °
Pitch	0.10
Bank	0.10

3.1.1.1.11.5 Servo Damping. The instrument servos shall be damped so the overshoot in response to a stop input shall not exceed 10 percent.

3.1.1.1.11.6 Self Test. A manually actuated self-test mode shall be provided to enable a preflight checkout of the attitude portion of the indicator. The attitude failure flag shall be in view during the self-test mode.

3.1.1.1.11.7 Failure Warning Requirements. An attitude failure warning flag shall be provided which shall become visible if any of the following conditions exist:

a. Loss of electrical power to the indicator
b. Failure of servo-power supply
c. Absence of an input valid signal
d. Excessive servo-null voltages

The flag signal shall be a dc voltage and shall be available for external use.

3.1.1.1.12 Approach Progress Annunciator. The functions of the approach progress annunciator shall be:

a. To inform the flight crew of the actual accomplishment of mode changes which take place during autopilot/flight director approach and landing operation.

b. To inform the flight crew of passing the minimum decision altitude during landing approach.

3.1.1.1.12.1 Method of Presentation and Mechanization. The display arrangement shall be essentially shown in Fig. 30. The display devices can be either indicator lights or integrally lighted flag type annunciators or a combination of both. The indicator segments shall be designed so the legends are not visible, under all ambient lighting conditions, when the indicators are deactivated. When activated, the legends shall be readily visible under all ambient lighting conditions except direct sunlight. Manual dimming shall not be required.

3.1.1.1.12.1.1 VOR/LOC Annunciator. The VOR/LOC annunciator shall appear amber when a signal is received from the automatic flight control system (AFCS) indicating it is in the VOR/LOC armed mode. The annunciator shall change to green when a signal is received from the AFCS indicating it has changed to the VOR/LOC capture mode.

3.1.1.1.12.1.2 GS Annunciator. The glide slope annunciator shall appear amber when a signal is received from the AFCS indicating it is in the GS armed mode. The annunciator shall change to green when a signal is received from the AFCS indicating it has changed to the GS capture mode.

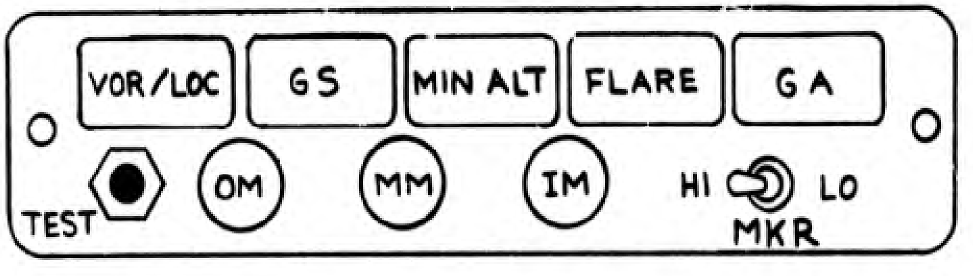


Figure 30. Approach Progress Annunciator

D6A10122-1

3.1.1.1.12.1.3 MIN ALT Annunciator. The minimum altitude annunciator shall flash amber when a minimum altitude signal is received.

3.1.1.1.12.1.4 Flare Annunciator. The flare annunciator shall appear amber when a signal is received from the AFCS indicating the auto-land mode is selected and operating. The annunciator shall change to green when a signal is received from the FACS indicating a flare has been commanded.

3.1.1.1.12.1.5 GA Annunciator. The go-around annunciator shall appear amber simultaneously with the flare annunciator appearing amber provided a signal is present indicating the AFCS go-around mode is operable. The annunciator shall change to green when a signal is received from the AFCS indicating go-around mode has been selected.

3.1.1.1.12.1.6 Marker Beacon Annunciators and Sensitivity Switch. The outer marker (OM) shall appear blue, the middle marker (MM) shall appear amber and the inner marker (IM) shall appear white when the respective signals are received from the marker beacon receiver. A switch shall provided to enable the selection of high-or-low sensitivity for the marker receiver.

3.1.1.1.12.1.7 Self Test. A self-test mode shall be provided. Operation of the self-test mode switch into two detent positions shall activate all indicator segments of the unit.

3.1.1.1.13 Instrument Warning System. The comparator, warning flag and approach gate monitor shall sense the following parameters and provide an aural and/or visual warning when the parameter deviates from a preset value by more than a specified amount.

a. The difference between flight and navigation information being displayed on the pilots' panels.

b. The individual warning flag signals.

c. Localizer and glide slope valid signals during the final phase of landing approach.

The monitor shall have the capability to sense and display warnings as required within the following channels of information:

a. Attitude
b. Heading
c. Barometric Altitude
d. Radio Altitude
e. Calibrated Airspeed
f. Glide Slope
g. Localizer
h. Flight Director Commands

3.1.1.1.13.1 Method of Presentation and Mechanization. The display unit arrangement shall be essentially as shown in Fig. 31. The display unit shall provide individual annunciators for an approach gate monitor channel, five comparison channels (Airspeed, Baro Altitude, Heading, Attitude and Command) and eight flag warning annunciator pairs (Airspeed, No. 1 and 2, Baro Altitude No. 1 and 2, Heading No. 1 and 2, Attitude No. 1 and 2, Command No. 1 and 2, Radio Altitude No. 1 and 2, Glide Slope No. 1 and 2, Localizer No. 1 and 2. A pushbutton switch shall be provided for actuating the self-test mode of the Control Unit.

3.1.1.1.13.2 Comparison Warning. When a comparison warning condition is detected, the master warning circuits and the appropriate comparison warning annunciator circuit(s) shall be activated. Operation of the reset switch shall remove the master warning signals and shall reduce the magnitude of the output voltage of the comparison warning annunciator circuit(s). If the fault condition is cleared, the comparison channel shall be re-armed and shall be capable of detecting a subsequent fault condition.

3.1.1.1.13.3 Flag Warning. When a flag warning condition is detected, the master warning circuits and the appropriate comparison warning annunciator circuit(s) shall be activated. Operation of the reset switch shall remove the master warning signals and shall reduce the magnitude of the output voltage of the flag warning and comparison annunciator circuits. If the fault condition is cleared, the flag warning channel shall be rearmed and shall be capable of detecting a subsequent fault condition.

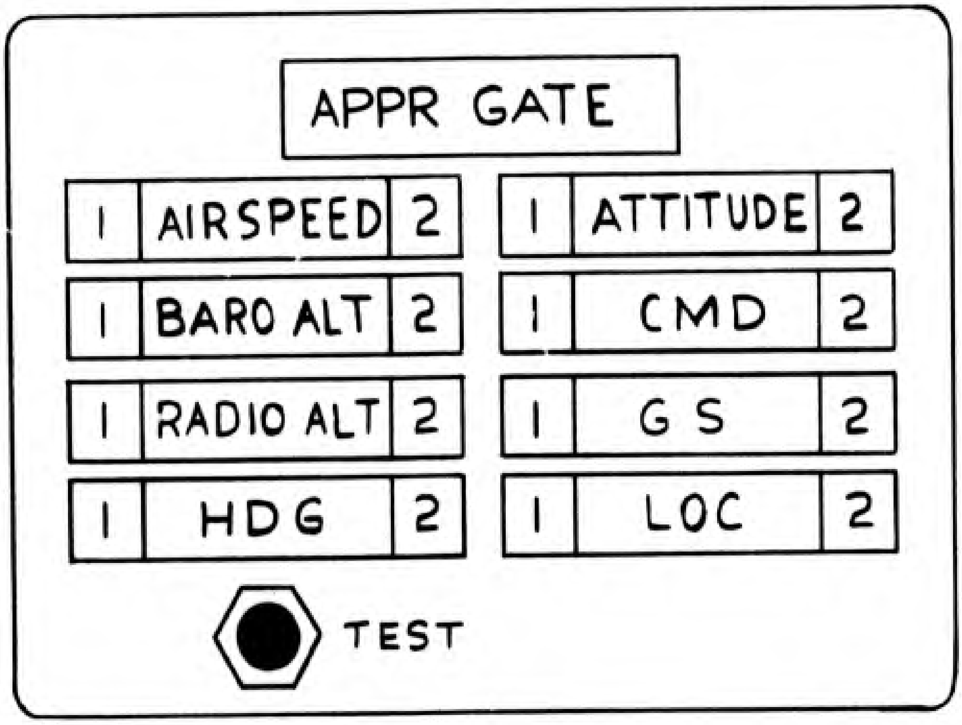


Figure 31. Instrument Warning Annunciator

D6A10122-1

3.1.1.1.13.4 Approach Gate Warning. When an approach gate warning condition is detected, the master warning circuit(s) and the approach gate warning annunciator circuit shall be activated. Operation of the reset switch shall remove the master warning signals and shall reduce the magnitude of the output voltage of the approach gate warning annunciator circuit.

3.1.1.1.13.5 Localizer Interlock. The No. 1 localizer flag warning channel shall be activated when the No. 1 localizer arming signal is present. The No. 2 localizer flag warning channel shall be activated when the No. 2 localizer arming signal is present.

3.1.1.1.13.6 Glide Slope Interlock. The No. 1 flight director flag warning channel shall be activated when the No. 1 flight director arming signal is present. The No. 2 flight director flag warning channel shall be activated when the No. 2 flight director arming signal is present. The flight director comparison channel shall be activated when both the No. 1 and No. 2 flight director arming signals are present.

3.1.1.1.13.7 Approach Gate Monitor Interlock. The approach gate monitor channel shall be activated when both the No. 1 and No. 2 glide slope arming signals are present. The glide slope portion of the approach gate monitor channel shall be deactivated at a radio altitude below 70 feet.

3.1.1.1.13.8 Warning Threshold. The signal levels corresponding to a fault condition shall be as defined in Table II. All comparison channels which operate on differential resolver input signals shall also monitor the excitation voltage (cosine output) of the resolver. A comparison warning shall be provided if the magnitude of the excitation signal falls below 50 percent of the normal level.

3.1.1.1.14 Stall Warning System Performance Requirements. The stall warning system shall actuate a warning device whenever the local angle of attack of the aircraft exceeds a value established by the flap position and wing-sweep angles.

3.1.1.1.14.1 Method of Mechanization. The stall warning system shall consist of the following components:

a. Two angle-of-attack sensors
b. Wing/Flap-position transmitter
c. Control Unit
d. Two control-stick shakers

The measured local angle of attack shall be sensed by the system along with the wing and flap positions. The system shall process this input information, compare it with pre-determined information standards and actuate a cockpit warning device(s) when the aircraft approaches a stalling situation.

3.1.1.1.14.1.1 Angle of Attack Sensor. The angle-of-attack sensor shall provide the means of measuring local air flow direction. The sensor shall have a range of ±25°.

Table II. Warning Threshold

Channel	Fault Level
Attitude Comparison	
Pitch Angle Differences	Adjustable between 2 to 6°
Bank Angle Differences	Adjustable between 2 to 6°
Attitude Flag Warning	
No. 1 Attitude Flag	18 volts dc
No. 2 Attitude Flag	18 volts dc
Heading Comparison	Adjustable between 2 to 6°
No. 1 Heading Flag	18 volts dc
No. 2 Heading Flag	18 volts dc
Baro Altitude Comparison	-1,000 to 10,000 ft: Adjustable between 50 to 100 ft 10,000 ft: Adjustable between 250 to 400 ft
No. 1 Baro Altitude Flag	18 volts dc
No. 2 Baro Altitude Flag	18 volts dc
Calibrated Airspeed Comparison	Adjustable between 8 to 12 knots
No. 1 Calibrated Airspeed Flag	18 volts dc
No. 2 Calibrated Airspeed Flag	18 volts dc
No. 1 Glide Slope Flag	18 volts dc
No. 2 Glide Slope Flag	18 volts dc
No. 1 Localizer Flag	18 volts dc
No. 2 Localizer Flag	18 volts dc
No. 1 Radio Altitude Flag	18 volts dc
No. 2 Radio Altitude Flag	18 volts dc
Flight Director Command Comparison	To be determined
No. 1 Flight Director Flag	18 volts dc
No. 2 Flight Director Flag	18 volts dc
Approach Gate:	
Glide Slope	1,000 to 200 ft radio altitude: Adjustable between 50 to 100 millivolts 200 to 70 ft radio altitude: Adjustable between 100 to 150 millivolts

Table II (Cont.)

Channel	Fault Level
Approach Gate (Cont.): Localizer	1,000 ft radio altitude: Adjustable between 30 to 70 millivolts 300 ft radio altitude: Adjustable between 20 to 40 millivolts. 0 to 200 ft radio altitude: Adjustable between 10 to 30 millivolts

3.1.1.1.14.1.1.1. Calibration Error. The electrical output of the sensor shall correspond to the local direction of airflow with a maximum error of ± 0.20° at calibrated airspeeds above 110 knots.

3.1.1.1.14.1.1.2 Resolution. The sensor shall respond to a 0.2° change in airstream direction at a calibrated airspeed of 90 knots. The sensor shall respond to a 0.1° change in airstream direction at calibrated airspeeds above 125 knots.

3.1.1.1.14.1.1.3 Time Constant. The time constant of the sensor shall not exceed 0.1 sec. when subjected to a calibrated airspeed of 110 knots.

3.1.1.1.14.1.1.4 Damping. The sensor shall be damped so the overshoot in response to a step change of 3° at an airspeed of 110 knots, shall not exceed 0.5°.

3.1.1.1.14.1.1.5 Anti-Icing. The sensor shall be electrically heated. The heaters shall be capable of removing accumulated ice, and maintaining an ice free condition sufficient for normal operation under icing conditions specified in FAR Part 25, Par. 25.1419, and ambient temperatures down to - 40°C. The sensor shall not be damaged by continuous application of heater power under still air conditions.

3.1.1.1.14.1.2 Wing/Flap Position Transmitter. Wing sweep and flap-position transmitters shall provide compensation signals for the stall warning system.

3.1.1.1.14.1.3 Control Unit. The control unit shall be designed with no moving parts. All switching (other than self-test actuation) and function generation shall be performed by solid state devices.

3.1.1.1.14.1.4 Control Stick Shakers. The control-stick shakers shall consist of a motor-driven eccentric mass. They shall be mounted directly on the control sticks and, when activated, shall apply a vibratory force to the stick. The force output shall be pounds at a frequency of cycles per second.

3.1.1.1.14.2 System Accuracy. The repeatability of the stall warning actuation point for any low-speed flight condition and airplane configuration shall be equivalent to a change in local angle of attack of degrees. This tolerance shall include sensing, transmission and system errors.

3.2 SUBSYSTEM DEFINITIONS.

3.2.1 Interface Requirements.

3.2.1.1 Schematic Arrangement. See Figs. 32, 33, 34, and 35 for instrument displays and sensors subsystem interface schematics.

3.2.1.2 Detailed Interface Definition. The instrument displays and sensors subsystem shall provide interface with appropriate ARINC and FAA ground systems.

3.3 DESIGN AND CONSTRUCTION.

3.3.1 Subsystem Design Features.

3.3.1.1 Weight. The weight of the article shall be a minimum consistent with the performance requirements and within the limits of sound design practices.

3.3.1.2 Sealing. Use of hermetic sealing of major subassemblies shall be avoided where possible.

3.3.1.3 Outline. ARINC 404, Air Transport Equipment Cases and Racking, shall be a requirement in the design of the size, dimensional tolerances, mounting provisions, cooling (if required), and electrical connectors for those items of equipment installed within an equipment rack. ARINC 408, Air Transport Indicator Cases and Mounting, shall be utilized in the design of the size, dimensional tolerances and mounting provisions where practical for those items of equipment installed on an instrument panel.

3.3.1.4 Lighting. The indicators shall be integrally white lighted.

3.3.1.5 Non-Reflective Coating. The indicators cover glass shall be coated with non-reflective material.

3.3.1.6 Amplifier. The indicators shall be provided with integral amplifiers.

3.3.2 through 3.3.11 — See Part I of this specification.

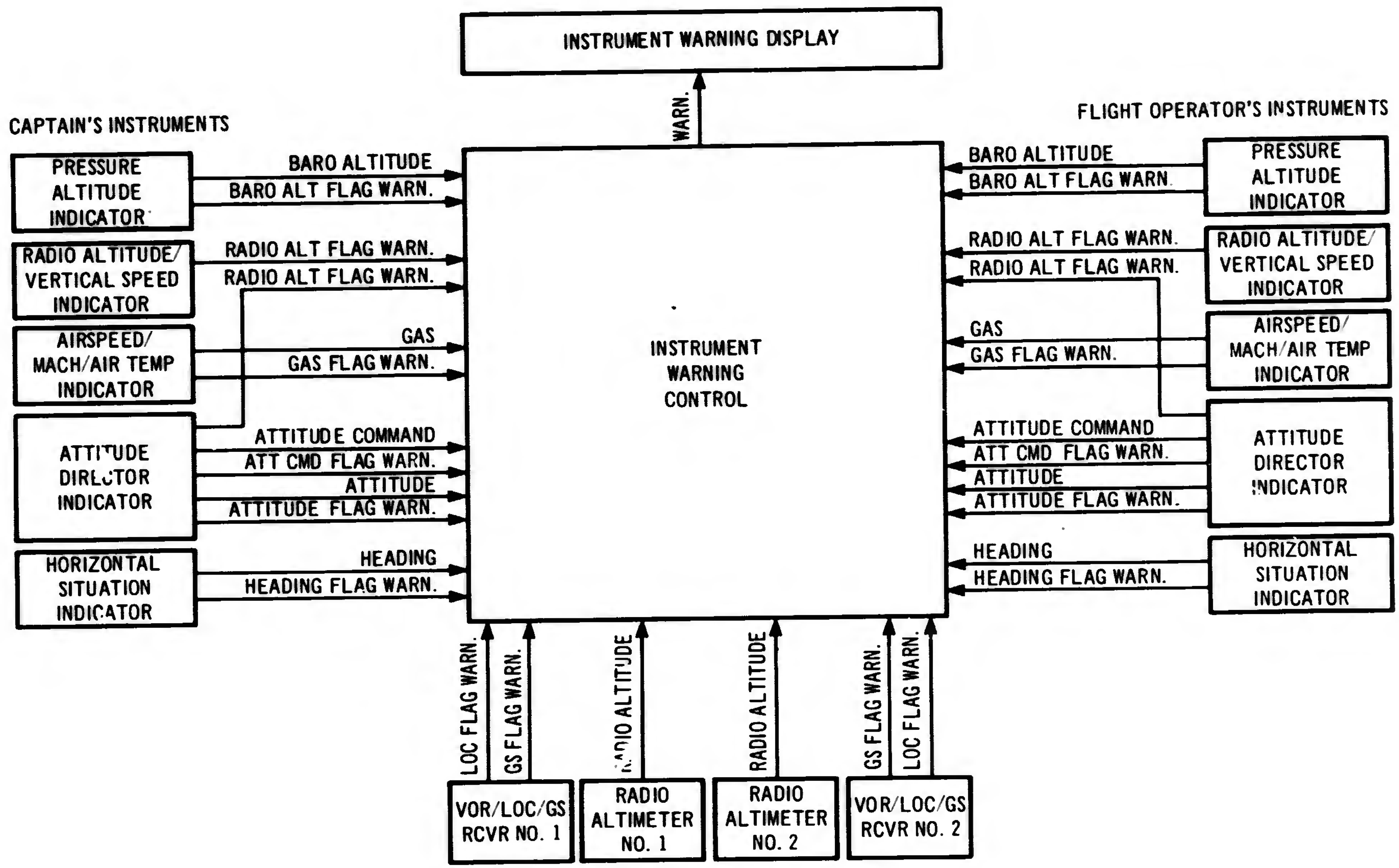


Figure 32. Instrument Warning System Interface Schematic

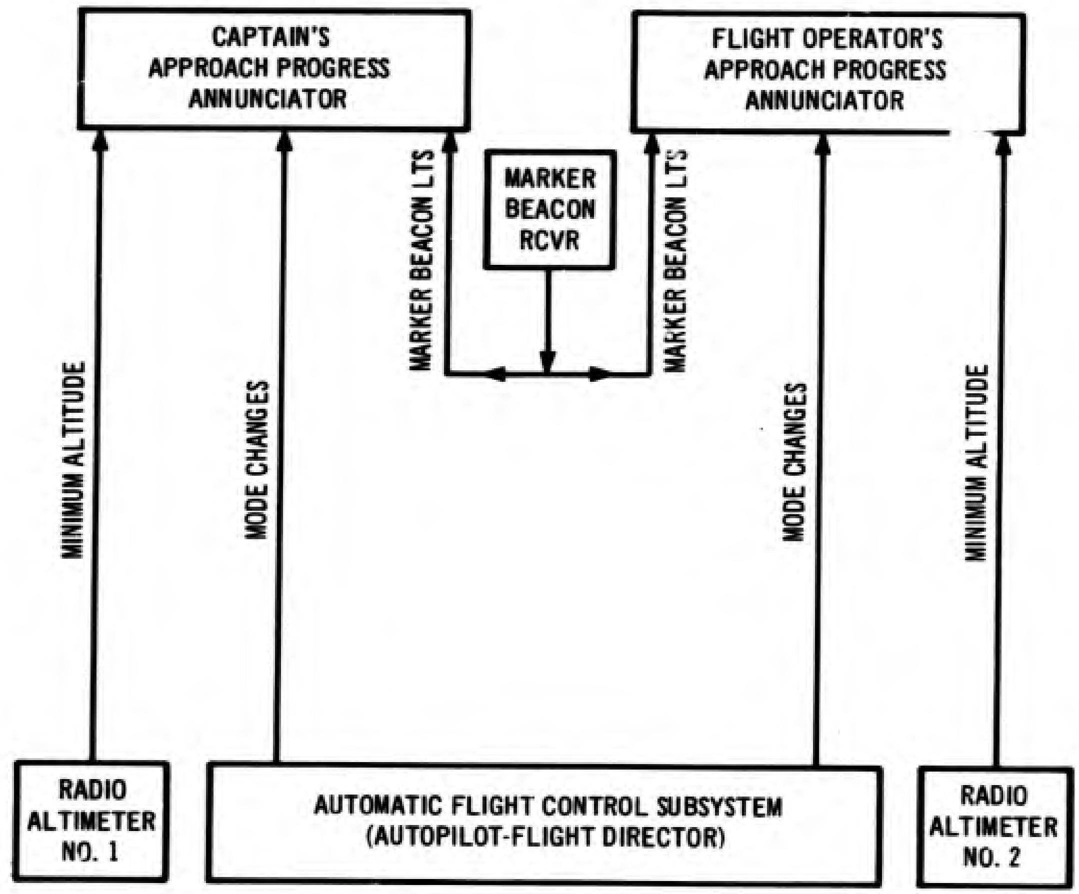
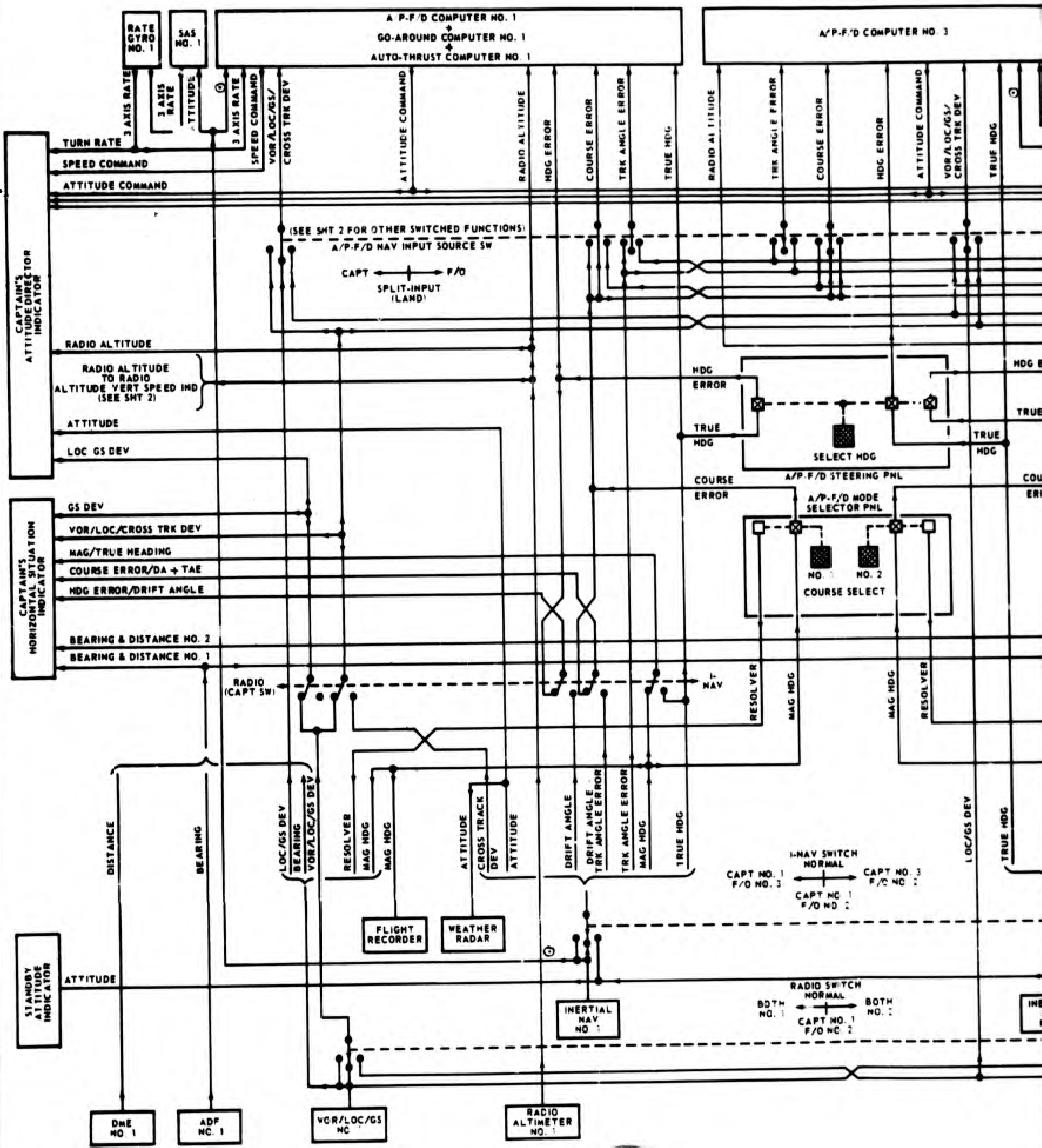


Figure 33. Approach Progress Annunciator Interface Schematic



RATE GYRO NO. 1
SAS NO. 1
A/P-F/D COMPUTER NO. 1 + GO-AROUND COMPUTER NO. 1 + AUTO-THRUST COMPUTER NO. 1
A/P-F/D COMPUTER NO. 3
3 AXIS RATE
3 AXIS RATE
ATTITUDE
3 AXIS RATE
SPEED COMMAND
VOR/LOC/GS/ CROSS TRK DEV
ATTITUDE COMMAND
RADIO ALTITUDE
HDG ERROR
COURSE ERROR
TRK ANGLE ERROR
TRUE HDG
RADIO ALTITUDE
TRK ANGLE ERROR
COURSE ERROR
HDG ERROR
ATTITUDE COMMAND
VOR/LOC/GS/ CROSS TRK DEV
TRUE HDG
CAPTAIN'S ATTITUDE DIRECTOR INDICATOR
TURN RATE
SPEED COMMAND
ATTITUDE COMMAND
(SEE SHT 2 FOR OTHER SWITCHED FUNCTIONS)
A/P-F/D NAV INPUT SOURCE SW
CAPT
F/O
SPLIT-INPUT (LAND)
RADIO ALTITUDE
RADIO ALTITUDE TO RADIO ALTITUDE VERT SPEED IND (SEE SHT 2)
ATTITUDE
LOC GS DEV
HDG ERROR
TRUE HDG
SELECT HDG
A/P-F/D STEERING PNL
TRUE HDG
COURSE ERROR
A/P-F/D MODE SELECTOR PNL
NO. 1
NO. 2
COURSE SELECT
CAPTAIN'S HORIZONTAL SITUATION INDICATOR
GS DEV
VOR/LOC/CROSS TRK DEV
MAG/TRUE HEADING
COURSE ERROR/DA + TAE
HDG ERROR/DRIFT ANGLE
BEARING & DISTANCE NO. 2
BEARING & DISTANCE NO. 1
RADIO (CAPT SW)
I-NAV
RESOLVER
MAG HDG
MAG HDG
RESOLVER
DISTANCE
BEARING
LOC/GS DEV
BEARING
VOR/LOC/GS DEV
RESOLVER
MAG HDG
MAG HDG
ATTITUDE
CROSS TRACK DEV
ATTITUDE
DRIFT ANGLE
DRIFT ANGLE - TRK ANGLE ERROR
TRK ANGLE ERROR
MAG HDG
TRUE HDG
LOC/GS DEV
TRUE HDG
I-NAV SWITCH NORMAL
CAPT NO. 1 F/O NO. 3
CAPT NO. 3 F/O NO. 2
CAPT NO. 1 F/O NO. 2
FLIGHT RECORDER
WEATHER RADAR
STANDBY ATTITUDE INDICATOR
ATTITUDE
INERTIAL NAV NO. 1
RADIO SWITCH NORMAL
BOTH NO. 1
BOTH NO. 2
CAPT NO. 1 F/O NO. 2
DME NO. 1
ADF NO. 1
VOR/LOC/GS NO. 1
RADIO ALTIMETER NO. 1

1

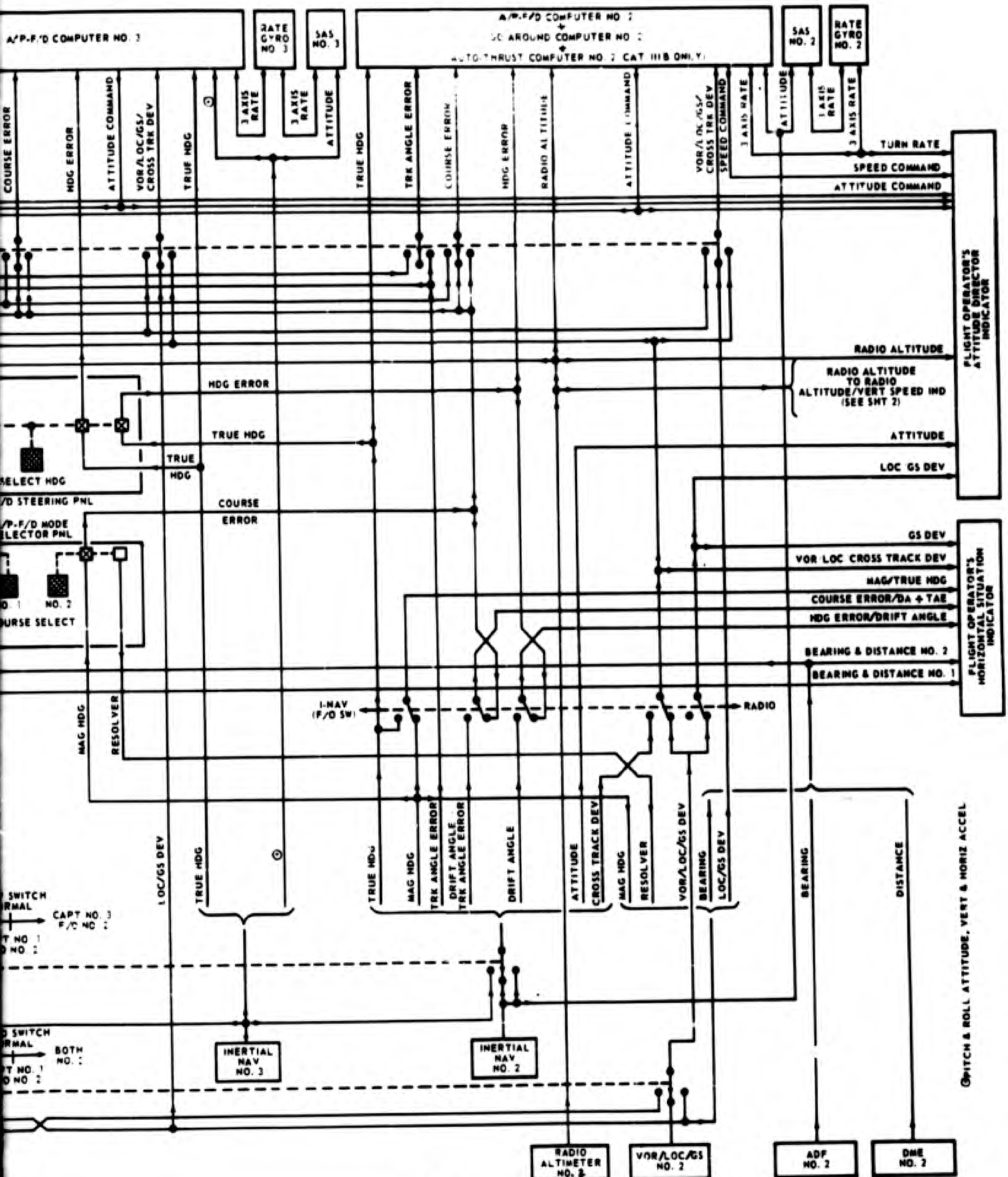
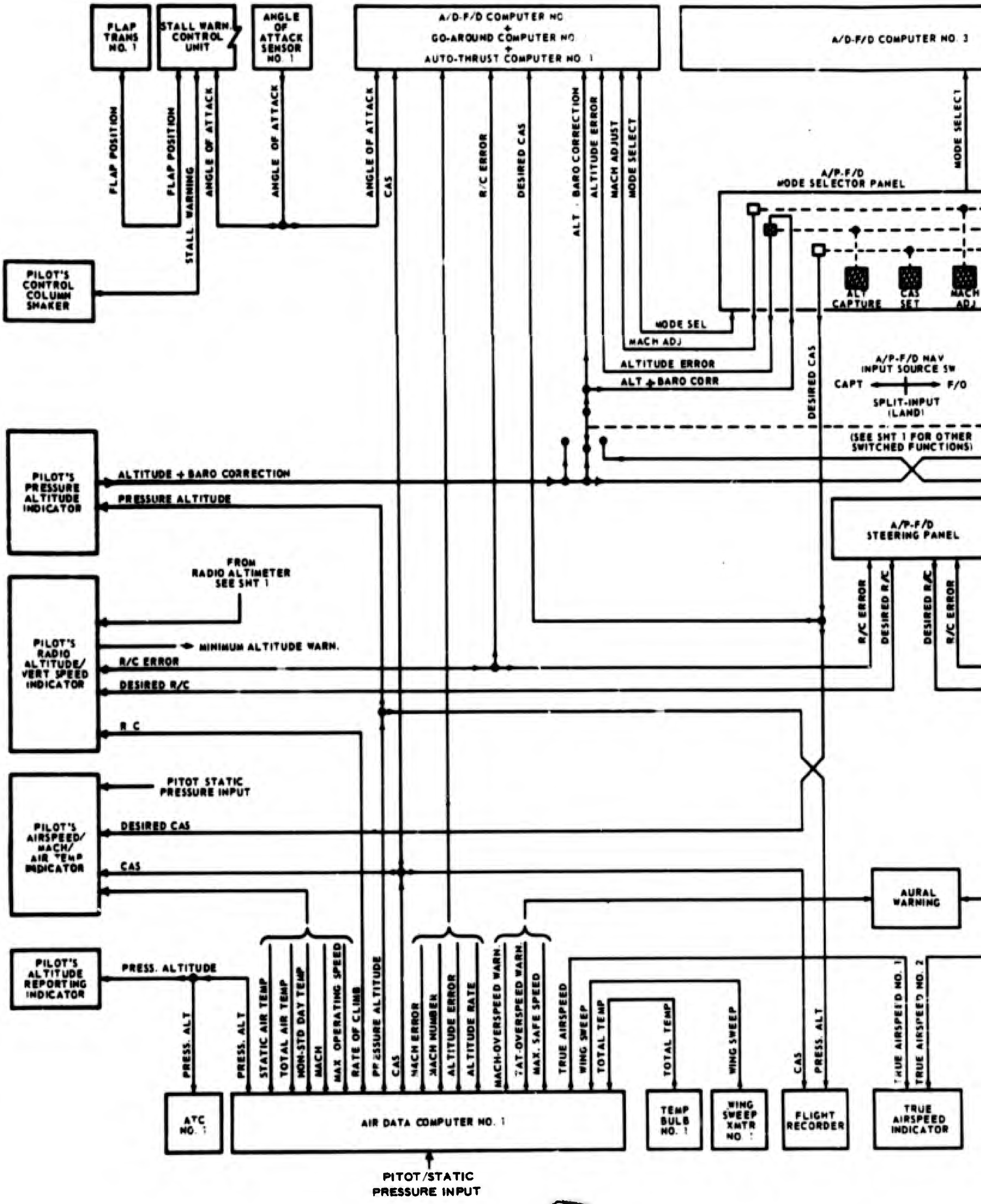


Figure 34. Navigation Systems Interface

2


FLAP TRANS NO. 1
STALL WARN. CONTROL UNIT
ANGLE OF ATTACK SENSOR NO. 1
A/D-F/D COMPUTER NO + GO-AROUND COMPUTER NO + AUTO-THRUST COMPUTER NO. 1
A/D-F/D COMPUTER NO. 3
FLAP POSITION
FLAP POSITION
STALL WARNING
ANGLE OF ATTACK
ANGLE OF ATTACK
ANGLE OF ATTACK
CAS
R/C ERROR
DESIRED CAS
ALT + BARO CORRECTION
ALTITUDE ERROR
MACH ADJUST
MODE SELECT
MODE SELECT
A/P-F/D MODE SELECTOR PANEL
ALT CAPTURE
CAS SET
MACH ADJ
PILOT'S CONTROL COLUMN SHAKER
MODE SEL
MACH ADJ
ALTITUDE ERROR
ALT + BARO CORR
DESIRED CAS
A/P-F/D NAV INPUT SOURCE SW
CAPT
F/O
SPLIT-INPUT (LAND)
(SEE SHT 1 FOR OTHER SWITCHED FUNCTIONS)
PILOT'S PRESSURE ALTITUDE INDICATOR
ALTITUDE + BARO CORRECTION
PRESSURE ALTITUDE
A/P-F/D STEERING PANEL
FROM RADIO ALTIMETER SEE SHT 1
R/C ERROR
DESIRED R/C
DESIRED R/C
R/C ERROR
PILOT'S RADIO ALTITUDE/ VERT SPEED INDICATOR
MINIMUM ALTITUDE WARN.
R/C ERROR
DESIRED R/C
R C
PITOT STATIC PRESSURE INPUT
PILOT'S AIRSPEED/ MACH/ AIR TEMP INDICATOR
DESIRED CAS
CAS
AURAL WARNING
PILOT'S ALTITUDE REPORTING INDICATOR
PRESS. ALTITUDE
PRESS. ALT
PRESS. ALT
STATIC AIR TEMP
TOTAL AIR TEMP
NON-STD DAY TEMP
MACH
MAX. OPERATING SPEED
RATE OF CLIMB
PRESSURE ALTITUDE
CAS
MACH ERROR
MACH NUMBER
ALTITUDE ERROR
ALTITUDE RATE
MACH-OVERSPEED WARN.
TAT-OVERSPEED WARN.
MAX. SAFE SPEED
TRUE AIRSPEED
WING SWEEP
TOTAL TEMP
TOTAL TEMP
WING SWEEP
CAS
PRESS. ALT
TRUE AIRSPEED NO. 1
TRUE AIRSPEED NO. 2
ATC NO. 1
AIR DATA COMPUTER NO. 1
TEMP BULB NO. 1
WING SWEEP XMTR NO. 1
FLIGHT RECORDER
TRUE AIRSPEED INDICATOR
PITOT/STATIC PRESSURE INPUT

1

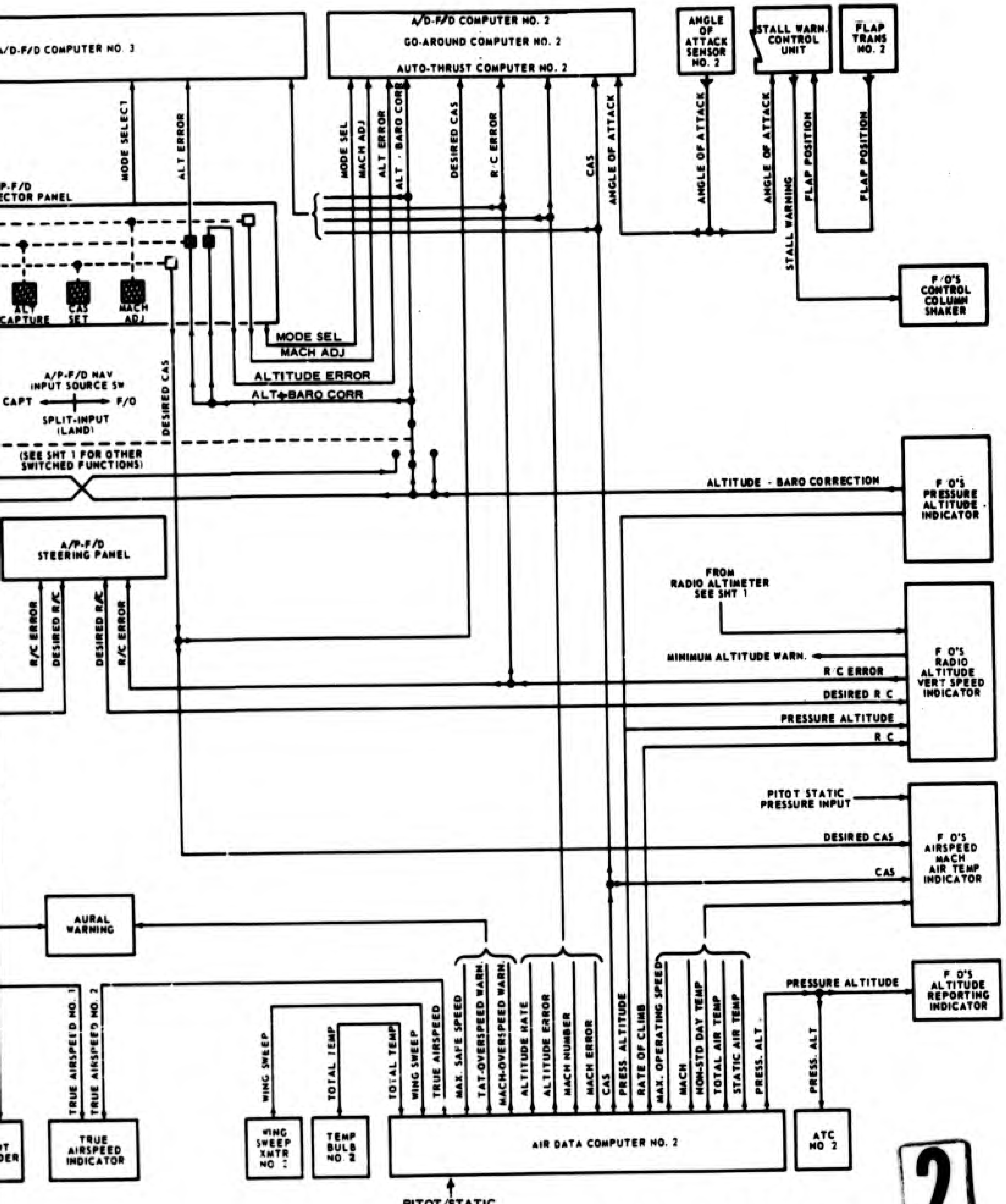


Figure 35. Air-Data System Interface

2

4.0 QUALITY ASSURANCE PROVISIONS

4.1 ENGINEERING TEST AND EVALUATION. Engineering tests and evaluations will be conducted as described in Par. 4.1 of Part I of this subsystem specification. Wind tunnel testing shall be utilized to determine the installation location of the pitot and static sources and will be performed to check pitot-static sensor repeatability and effects of changes in airflow direction on measured pressure. Tests shall be conducted in an icing tunnel to establish compliance of icing protection requirements of the total temperature probe, angle-of-attack sensor and pitot-static source.

4.2 PRELIMINARY QUALIFICATION TESTS. Refer to Par. 4.2 of Part I.

4.3 FORMAL QUALIFICATION TESTS. Refer to Par. 4.3 of Part I.

4.3.1 Inspection. Inspection will be conducted as described in Par. 4.3.1 of Part I of this subsystem specification. Inspection of all displays shall be conducted to verify the method of presentation and mechanization including display markings, lighting and non-reflective cover glass coating.

4.3.2 Analysis. Refer to Par. 4.3.2 of Part I.

4.3.3 Demonstration. Refer to Par. 4.3.3 of Part I.

4.3.4 Tests. Refer to Par. 4.3.4 of Part I.

a. Subsystem Integration Tests. Subsystem integration tests will be conducted as described in Par. 4.3.4.a of Part I of this subsystem specification and shall be conducted on a component basis and an integrated system basis. Component testing shall consist of supplying simulated test equipment inputs to each function over its specified range. The outputs including display presentations shall be monitored for verification of performance requirements.

   The instrument displays and sensors shall be interconnected as shown in Figs. 32, 33, 34, and 35 for an integrated system test. The display functions shall be monitored in conjunction with the testing conducted on the associated navigation subsystem test. Failure monitoring and self-test features shall be demonstrated and verified.

b. Instrument displays and sensors shall be ground and flight tested for adequacy of performance.

   Ground testing will be a duplication of the test procedures developed in Par. 4.3.4.a and will verify the results found therein. Leakage tests of the pitot-static system shall be conducted.

   Flight tests shall be conducted to calibrate the pitot-static system, total air temperature probe and stall warning system. Compatibility of the air-data system and autopilot system shall be demonstrated. Satisfactory operation of fault detection and failure warning shall be demonstrated. Adequacy of display performance, presentation and arrangement for operation of airplane over the complete flight profile including all-weather landing shall be demonstrated.

4.4 RELIABILITY TEST AND ANALYSIS. Refer to Par. 4.4 of Part I.